USAAEFA PROJECT NO. 81-11

AD-A140 986

JUH-1H PNEUMATIC BOOT DEICING SYSTEM FLIGHT TEST EVALUATION

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FINAL REPORT

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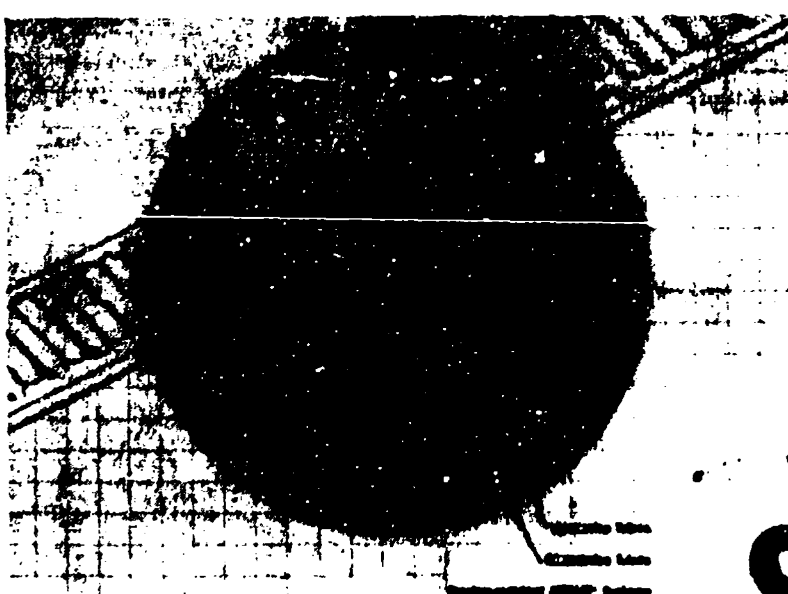

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Artificial Icing Tests

were conducted

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The concept of operation of the Pneumatic Boot Deicing System as a deice system is feasible under the conditions tested when installed on the test UH-1H helicopter. A cooperative effort between NASA and the B.F. Goodrich Company led to the development of a prototype pneumatic boot deicing system designed for use on a helicopter main rotor system. The US Army Aviation Engineering Flight Activity was tasked to conduct flight tests to establish an operational envelope and evaluate the feasibility of the pneumatic deicing concept. ~~The testing~~

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~~was conducted in three~~ phases between 16 November 1981 and 30 May 1983. Phase I consisted of a ground and inflight structural loads survey which established an operational envelope. Phase II was a limited aircraft performance and handling qualities evaluation, and Phase III involved artificial icing tests. Thirteen ~~(13)~~ problem areas were identified with the prototype Pneumatic Boot Deicing System installation, two of which were corrected during the evaluation. Major problem areas included excessive increases in power required for flight and deicer material erosion and breakdown.

DEPARTMENT OF THE ARMY
HEADQUARTERS, US ARMY AVIATION SYSTEMS COMMAND
4300 GOODFELLOW BOULEVARD, ST. LOUIS, MO 63120

REPLY TO
ATTENTION OF

DRSAV-E

SUBJECT: Directorate for Engineering Position on the Final Report of USAAEFA Project No. 81-11, JUH-1H Pneumatic Boot Deicing System Flight Test Evaluation

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1. The purpose of this letter is to establish the Directorate for Engineering position on the subject report. The objective of this evaluation was to determine the feasibility of a Pneumatic Boot Deicing System (PBDS) on helicopter rotor blades. The UH-1H was selected as a viable helicopter on which to conduct the evaluation mainly due to its suitable configuration and availability. The evaluation was a joint effort by the US Army, National Aeronautics and Space Administration (NASA) and BF Goodrich (BFG) Company. There was no intent to conduct an Airworthiness Qualification Program (AQP) per AR 70-62 since fielding of the UH-1H configured with a PBDS was not a requirement. All Airworthiness testing conducted during the evaluation was strictly for releasing an envelope such that the PBDS could be adequately evaluated. It should be noted that a carefully controlled three phase program was conducted which culminated in artificial icing tests using the Icing Spray Rig at Ottawa, Canada.

2. This Directorate agrees with the report conclusions and recommendations with the exceptions identified herein. Comments are directed to the paragraph of the report as indicated below:

a. Paragraphs 60a., d., e., f., g., i., and j. These paragraphs relate to recommending changes to the PBDS prior to further testing. While the changes are desirable and encouraged, they are not mandatory and should be left to NASA/BFG to decide prior to icing tests being conducted in forward flight.

b. Paragraphs 60b., c., k., l., n., o., and p. These paragraphs relate to recommended changes to the PBDS prior to US Army type operational testing. While these changes would be necessary prior to operational testing they are not mandatory for continued research and development testing of the PBDS.

c. Para 60m. The recommendation to reduce airspeed to below 80 KCAS and land as soon as practical following a PBDS failure in the vented condition will be incorporated in future operations. This is necessary to reduce the severe vertical vibration encountered with the failed PBDS.

DRSAV-E
SUBJECT: Directorate for Engineering Position on the Final Report of USAAEFA Project No. 81-11, JUH-1H Pneumatic Boot Deicing System Flight Test Evaluation

d. Para n. The recommendation to reduce spanwise coverage of the PBDS should be a prime effort. This may significantly reduce profile drag and significantly improve the overall PBDS performance.

3. The report prepared by the US Army Aviation Engineering Flight Activity (USAAEFA) provides excellent documentation of a well executed research flight test program tc date. The problem areas which if corrected by NASA/BFG and reevaluated by USAAEFA could lead to a lightweight and cost effective helicopter rotor blade deicing system. However, it should be noted that the flight performance loss with the PBDS operating is excessive compared to elecrical deicing systems. A final consideration is one related to the type of airfoil tested with the PBDS concept installed. The test JUH-1H incorporates a NACA 0012 rotor blade while newer airfoil sections such as the SC1095R8 are aerodynamically advanced. The new airfoils could result in significantly different PBDS problems as compared to the 0012 airfoil PBDS installation and these differences should be considered for any future testing. The next phase of JUH-1H PBDS testing should be conducted in forward flight under artificial and natural icing conditions. This is considered essential to the research effort since forward flight represents the normal mode of flight where different problems associated with the PBDS will be identified.

RONALD E. GORMONT
Acting Director of Engineering

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INTRODUCTION

BACKGROUND

1. Pneumatic deicer boots have seen extensive application on fixed wing aircraft. Numerous material problems, however, have prevented their use on helicopter rotors. The concept was revived by development of an erosion resistant polyurethane elastomer (trade name ESTANE®) by the BF Goodrich (BFG) Company. In 1979, the National Aeronautics and Space Administration's (NASA) Lewis Research Center in cooperation with BFG conducted limited wind tunnel icing tests to evaluate the feasibility of pneumatic boot deicing concepts for helicopter rotor systems as reported by NASA Lewis in 1980 (ref 1, app A). To flight test the concept, NASA's Ames Research Center requested the assistance of the US Army Aviation Research and Development Command (AVRADCOM) (ref 2, app A). AVRADCOM subsequently tasked the US Army Aviation Engineering Flight Activity (USAAEFA) to conduct flight tests to evaluate the feasibility of the pneumatic concept for helicopter rotor systems (ref 3, app A). A test plan (ref 4, app A) was submitted in September 1981 and revised in March 1982. The airworthiness release (ref 5, app A) was initially issued 18 November 1981.

2. BFG manufactured the prototype ESTANE® pneumatic boots and installed them on a set of government-furnished UH-1H main rotor blades provided by NASA-Ames. BFG also furnished the de-icer system control components. Bell Helicopter Textron (BHT), under a NASA contract, instrumented two main rotor blades, main rotor hub, mast, pitch change link, and the three main rotor control actuator extension tubes for inflight measurement of structural loads. BHT also designed and built an instrumentation and pneumatic slip ring assembly (incorporating the BFG furnished rotating union) which was installed on the test helicopter at USAAEFA prior to testing.

TEST OBJECTIVES

3. The objectives of this evaluation were to determine an operational flight envelope and conduct feasibility testing of the Pneumatic Boot Deicing System (PBDS) conception on a JUH-1H in both dry air and artificial icing conditions.

DESCRIPTION

4. The test JUH-1H is a thirteen-place single engine helicopter using a single two-bladed teetering main rotor and a two-bladed pusher tail rotor. The maximum gross weight of the helicopter is 9500 pounds. Power is provided by a Lycoming T53-L-13 free turbine engine rated at 1400 shaft horsepower (SHP). However,

the helicopter is limited by the transmission to 1100 SHP. A more complete description may be found in the operator's manual (ref 6, app A). The test JUH-1H helicopter, USA S/N 70-16318, was a standard production helicopter manufactured by BHT and has been modified to incorporate a partial ice protection sytem (Kit A) described in reference 7, appendix A, a rotor brake, and the PBDS.

5. A prototype ESTANE® deicer boot was applied to the leading edge of a standard UH-1H rotor blade, as shown in figure A. The deicer boot is designed to remove accumulated ice through pneumatic expansion (inflation) of chordwise and spanwise tubes.

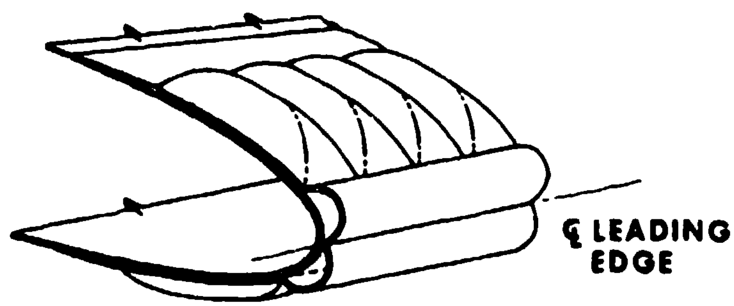


Figure A. Typical Cross Section of Installed Deicer (Inflated)

The PBDS also consists of a modified mast, electrical and pneumatic slipring, associated controllers, electrical components, and air supply components for providing engine bleed air to the PBDS as shown in figure B. Customer bleed air from the engine is routed to the deicers in a single inflation activation cycle. A normal activation cycle consists of inflation of the deicer boots for approximately two seconds, followed by subsequent deflation. A more detailed description of non-standard features of the test aircraft and the pneumatic deicing system is contained in appendix B.

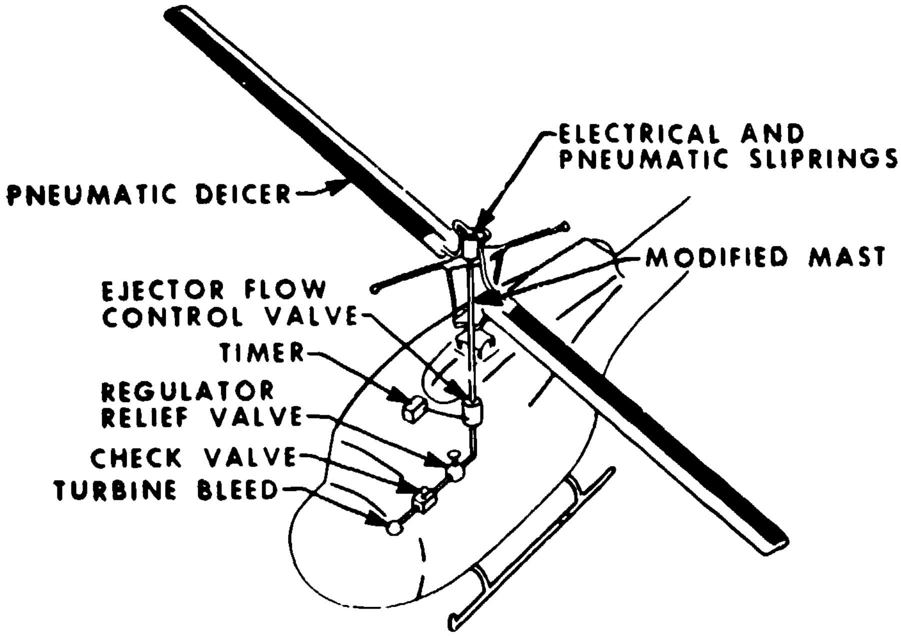


Figure B. Pneumatic Deicer System Components

TEST SCOPE

6. PBDS testing was conducted in three phases. The flight loads survey (Phase I) and performance and handling qualities testing (Phase II) were conducted at Edwards Air Force Base (EAFB), California during the period of 16 November 1981 through 30 Oct 1982. Artificial icing tests (Phase III) were conducted between 4 January and 25 February 1983 at Ottawa, Ontario, Canada. A total of 41 test flights were conducted during which 39.6 hours were flown. Where possible, flight test data were compared with data obtained from previous flight testing of the UH-1H (refs 8, 9 and 10, app A).

7. USAAEFA had overall responsibility for conduct of the test to include maintenance and instrumentation of the test helicopter. BHT and BFG furnished technical assistance during the installation of the system. NASA-Ames provided funding for the evaluation and technical and engineering support as required. AVRADCOM and BHT provided a dynamicist to monitor structural loads during all test phases.

8. The test aircraft was operated within the limitations of the operator's manual as amended by the airworthiness release (ref 5, app A) and limitations found during developmental testing of the PBDS. Flight tests were conducted at the test conditions shown in table 1.

TEST METHODOLOGY

9. Established flight test techniques (refs 11, 12, and 13, app A) were used throughout this evaluation where possible. Established test methods used are briefly discussed in the Results and Discussion section of this report while other nonstandard test methods are fully discussed in appendix D with a brief discussion in the Results and Discussion section. The Handling Qualities Rating Scale (HQRS) shown in figure 1, appendix D, and the Vibration Rating Scale (VRS) shown in figure 2, appendix D, were used to supplement pilot's qualitative comments. Flight test data were recorded by hand and on magnetic tape. Six rotor system component loads were monitored via telemetry (TM) to verify that loads were within allowable limits. A detailed listing of test instrumentation is contained in appendix C. Data analysis methods used are presented in appendix D. UH-1H Control System Loads monitoring trail is presented in appendix F.

Table 1. Test Conditions[1]

Test	Average Gross Weight (lb)	Average Density Altitude (ft)	Trim Calibrated Airspeed (knots)	Remarks
Loads Survey	7120 to 9500	1800 to 7600	0 to 100	Inflight and ground structural testing
Hover Performance	7420 to 7640	1900	0	In-ground effect (5-foot skid height), tethered method
Level Flight Performance	8140, 7880	8020, 9160	36 to 98	Ball centered, constant W/σ test technique
Autorotational Qualitative Evaluation	7900	6500	70, 80, 90	Ball centered collective full down
Control Positions in Trimmed Forward Flight	7800 to 8140	5120 to 9160	36 to 98	
Low-Speed Flight	7640 to 7800	3220	0-20[2] rearward 0-40[2] forward 0-20[2] left & right	5-foot skid height
Spray Rig Icing	7160 to 8000	-4700 to -2620	5 - 15[2]	Hover in spray rig cloud. 5 to 40-foot skid height.

NOTE:

[1]Normal Utility Configuration, mid cg.
[2]Knots true airspeed

RESULTS AND DISCUSSION

GENERAL

10. The evaluation of the PBDS consisted of three phases. Phase I was a ground and inflight structural loads survey which established an operational envelope up to 100 knots calibrated airspeed (KCAS) in forward flight (deflated configuration) and 20 knots true airspeed (KTAS) in sideward and rearward flight. Monitored structural loads demonstrated that the test helicopter with the PBDS installed may be safely flown through the established envelope. Six problems with the PBDS were discovered during this phase, three of which were subsequently corrected by BFG. Phase II was a limited aircraft performance and handling qualities evaluation. The installation of the PBDS (deflated configuration) adversely affected the hover and level flight performance of the UH-1H, with large increases in power required for flight. Handling qualities were not affected except for activation of the pneumatic deicer resulting in a right aircraft yaw. Phase III involved artificial icing tests in the National Research Council of Canada (NRC) icing spray rig. These tests demonstrated that the PBDS removed ice from the main rotor blades. Deicer material erosion and material failures were documented during all test phases. Six additional problem areas were identified during Phase II and III testing.

PHASE I STRUCTURAL LOADS SURVEY

General

11. Loads survey tests were conducted to establish a limited flight envelope for the UH-1H helicopter with the PBDS installed and were evaluated at the test conditions listed in table 1. While a normal PBDS activation cycle is transient and consists of a single deicer boot inflation immediately followed by deflation, for test purposes three sustained configurations were used: (1) deflated, representing a condition prior to PBDS activation. (2) inflated, representing the pneumatically expanded boot condition (normally of short duration) reached just after PBDS activation, and (3) vented, representing a failure mode caused by loss of bleed air allowing the boots to inflate partially. The main rotor system was instrumented with strain gauges by BHT, who provided a structural dynamicist to monitor loads via TM to verify that loads during each test point were within allowable limits and would probably remain so during the next test point. PBDS temperatures were monitored at selected locations as described in appendix C to determine if temperatures were in excess of system component qualification values. As reported by the BHT dynamicist, measured loads demonstrated that the test

helicopter with the BFG PBDS installed may be safely flown through the established flight envelope. No unusual or unexpected dynamic responses of the main rotor or control systems due to the PBDS installation or activation were noted. There were six PBDS problem areas found during Phase I: (1) excessive pneumatic boot deflation time during blade rotation, (2) debonding of the pneumatic boot from the main rotor blade, (3) self-activation of the PBDS timer in a laboratory vibration test, (4) rupture (blowout) of the pneumatic boot during the inflation cycle, (5) pneumatic deicer boot erosion and, (6) reduction of pressure delivered to the pneumatic deicer boot through the heat soaked ejector control valve. All the problem areas except erosion resistance were improved or corrected during Phase I testing.

Ground Structural Tests

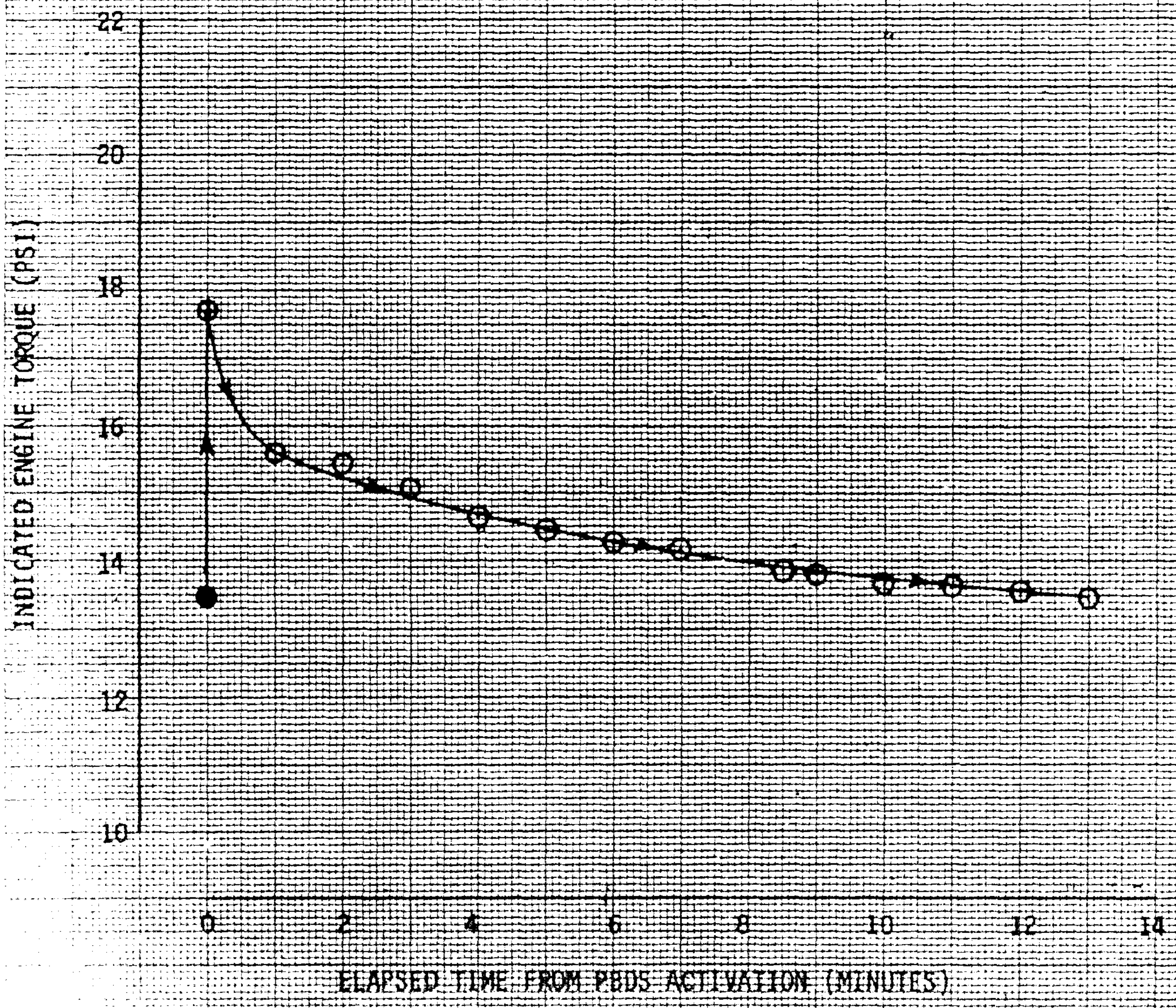
12. Ground tests were conducted at a 9500 pound gross weight with the aircraft secured by a clevis to a tie-down anchor. Rotor and pitch change link loads were monitored for each boot condition during start and run up to ground idle and 294 and 324 rotor revolutions per minute (rpm). After a few seconds at each rpm, the engine was shut down and the rotor system inspected. Once safe operation was established for rotor speeds of 294 and 324 rpm, engine power was incrementally increased to maximum at these rotor speeds. All loads monitored during the ground tests remained below endurance limits.

13. Dynamic system/engine compatability tests were done for 294 and 324 rotor rpm at three power settings (minimum, midrange, and maximum) by cycling the collective control and anti-torque pedals at a BHT determined critical frequency of the dynamic system (2.9 Hz) and 0.1 Hz above and below it. For each power setting at 324 rotor rpm the boots were inflated and deflated during an activation cycle and throttle reductions were made. No unusual dynamic response was noted throughout these tests.

14. During the ground runs, an activation cycle of the PBDS at flat pitch resulted in an engine torque increase of 3 pounds per square inch (PSI) over the baseline 13 PSI required to maintain constant rotor speed. Elapsed time required for torque to return to the baseline (deicer boot deflation time) was 12 to 14 minutes (fig. C). By comparison, with the main rotor blades static and the PBDS connected to shop air at the test connection, a complete cycle (inflation and deflation) of the deicer boots took 7 seconds. The deicer deflation time of 12 to 14 minutes was identified as a problem area and was decreased to 40 to 50 seconds by BFG after three modifications to the internal

FIGURE C
TIME HISTORY OF PNEUMATIC BOOT INFLATION/DEFLATION
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM

NOTES: 1. POWER TURBINE OUTPUT SHAFT SPEED = 6600 RPM
2. COLLECTIVE FULL DOWN
3. CYCLIC AND PEDALS HELD FIXED AT NEUTRAL
4. DARKENED SYMBOL DENOTES TRIM



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design of the pneumatic deicer boot which allowed progression to inflight testing. Excessive pneumatic deicer deflation of 40 to 50 seconds remains a problem area. Further reduction of deflation time should be accomplished prior to natural icing tests.

15. Two pneumatic deicer modifications required complete removal and installation of the deicer by BFG personnel. Total time required for retrofit by skilled BFG employees was approximately 16 man hours. On one occasion after retrofit, suspected contamination of the adhesive resulted in debonding of the pneumatic boot from the main rotor blade, as shown in photo 1, creating a problem area. The debonded pneumatic deicers were stripped from the main rotor blades and an additional set of deicers installed. No further deicer debonding of this type occurred during PBDS testing.

16. Temperatures of PBDS components during ground tests were monitored at selected locations described in appendix C by heat-sensitive tape and thermocouples to determine if temperatures created by the customer bleed air were in excess of component's qualification values. Heat-sensitive tape placed at the pressure inlet of the PBDS ejector flow control valve indicated a value of 160°F after system operation at high aircraft power settings (above 40-PSI torque). Initial BFG high-temperature environmental testing qualified the ejector valve to 160°F. The PBDS ejector flow control valve capability to pressurize the pneumatic deicer to 25 PSI decreased to as low as 16 PSI after the valve heat-soaked during normal deicer operation. The inability of the ejector flow control valve to pressurize the pneumatic deicer to the BFG-determined nominal pressure of 25 PSI was identified as a problem area. The ejector control valve was returned to BFG and was environmentally retested to 210°F in accordance with the experimental airworthiness release requirement to test to a temperature of 50°F greater than the maximum temperature recorded. Modification of the ejector control valve resulted in a higher deicer pressurization value of 22 PSI after the ejector control valve heat-soaked. Reduction of system pressure after the ejector control valve heat-soaks remains a problem area. Further modification of the ejector control valve to achieve the nominal system pressure of 25 PSI after heat-soak should be accomplished prior to operational testing.

17. Environmental testing of the PBDS timer was conducted at NASA-Dryden during the PBDS ground test phase. The timer was found to self-activate while undergoing vibration test in the longitudinal axis at 1.5 g's between 87 - 94 Hz and would not meet NASA specifications for flight assurance testing and specifications for

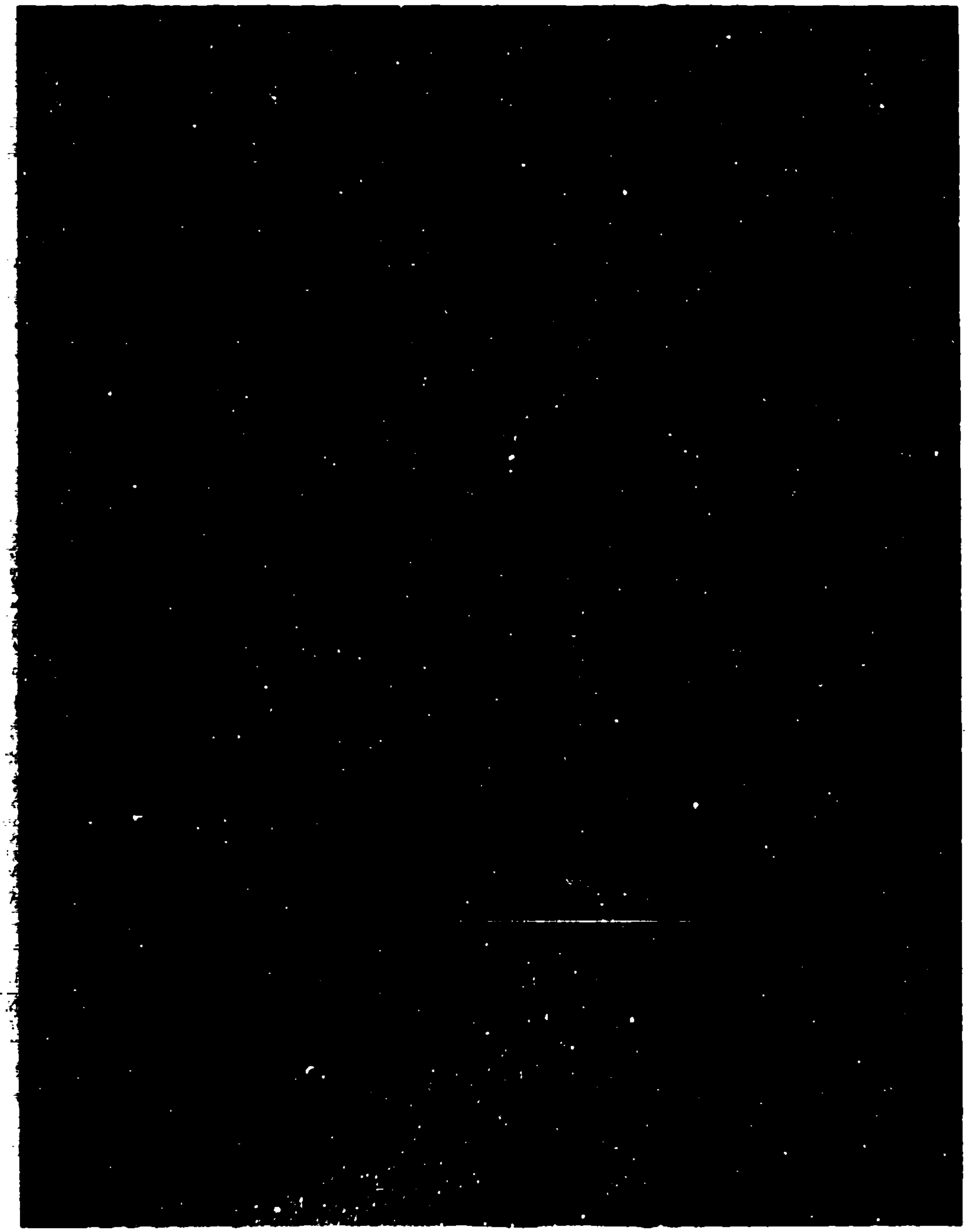


Photo 1. Deicer Debonded from Blade Surface

electrical systems (refs 14 and 15, app A). Timer self-activation was identified as a problem area. After modification of the timer by BFG, the modified timer was returned to NASA-Dryden for environmental testing. No failure modes were noted during vibrational tests of the modified timer, but eight further timer discrepancies were identified by the NASA-Dryden Safety and Quality Assurance Office. After correction of these discrepancies by BFG, the timer was returned to USAAEFA for use during PBDS testing. No further timer problems were encountered during flight testing.

Inflight Structural Tests

18. Inflight structural tests were conducted during hover, low airspeed, and forward flight at an 8000 pound takeoff gross weight and rotor speed of 324 rpm. Hover and low airspeed tests to 40 KTAS forward and 20 KTAS sideward and rearward were conducted in-ground effect (IGE) at all three deicer boot configurations (inflated, deflated and vented). Dynamic system-engine compatibility tests at a hover included collective control and anti-torque pedal inputs and deicer boot inflations similar to those of the ground tests. Forward flight testing involved PBDS activation cycles as well as the deflated and vented configurations. These tests included level flight from 50 to 100 KCAS, climbs and descents at 60 and 90 KCAS with engine power varied between minimum and maximum available, turns at 90 KCAS up to 1.6 g's, and autorotational entries at 70, 80 and 90 KCAS.

19. The only monitored load which exceeded endurance limits during the inflight structural test was the main rotor pitch link axial load. This load exceeded the structural endurance limit by 20% during activation of the PBDS in a 30° right coordinated turn and was not considered critical or unusual by the BHT dynamicist. There was no unusual dynamic response noted throughout these tests.

20. During hover and low airspeed structural tests, the outboard, leading edge area (beyond blade station 240) of the pneumatic deicer exhibited erosion and small punctures of the ESTANE® material. Erosion of the ESTANE® material was identified as a problem area and is discussed in the reliability and maintainability section (para 52) of this report.

21. While undergoing the PBDS operational check after installation of the modified ejector control valve, one pneumatic deicer boot ruptured (blow out). This created a slight rotor vibration with loss of system vacuum during deflation. Rupture of the pneumatic

deicer was recognized as a problem area and is discussed under the reliability and maintainability section (para 53) of this report.

PHASE II PERFORMANCE AND HANDLING QUALITIES

Performance

General:

22. Hover, level flight, and autorotational descent performance testing were conducted at EAFB, California (2302 foot elevation) at the conditions listed in table 1. Performance data were obtained with the PBDS in the deflated, vented, and inflated configurations described in paragraph 11. The sustained inflation configuration was used only for hover performance evaluation purposes and is not a normal failure mode of the PBDS. The performance data obtained with the PBDS installed was compared to standard UH-1H data in reference 8, appendix A.

Hover Performance:

23. Hover performance with deicer boots in the deflated, inflated, and vented configurations was evaluated using the tethered hover technique at a 5-foot skid height in winds of less than 3 knots. Figure 1, appendix E compares nondimensional hover performance for each boot configuration with standard-blade hover data from reference 8, appendix A. In dimensional terms, the performance penalties caused by the PBDS on a UH-1H at 9000 pounds on a standard day at sea level would be 100 SHP deflated, 230 SHP vented, and 370 SHP inflated. These values would correspond to power required increases of 11, 25, and 41 percent, respectively. The vented boot represents a condition that would exist with loss of bleed air; however, the inflated-boot performance data in figure 1 represents a transient condition found during inflation of the deicer boot during an activation cycle. The increase in power required to hover at 5 feet with the PBDS installed is a problem area which should be corrected in future pneumatic deicer designs.

Level Flight Performance:

24. The level flight performance of the test UH-1H with the PBDS installed was evaluated during trimmed ball-centered level flight at a thrust coefficient (C_T) of 0.0036 in the deflated and vented configurations using the constant gross weight-to-density ratio (w/δ) method. Test results are compared in figure 2, appendix E with the standard-bladed UH-1H data from reference 8,

appendix A. The PBDS was activated during steady state level flight at several airspeeds to evaluate the effect on the performance of the helicopter. A representative time history of the activation cycle at 81 KCAS is presented in figure 3, appendix E.

25. At 90 KCAS, approximately 170 additional SHP over standard UH-1H blades is required with PBDS deflated. With the PBDS vented, the additional SHP over standard blades increases to 340 SHP at 90 KCAS. The increase in power required for level flight with the PBDS installed is a problem area. The increase in power required for level flight with PBDS installed should be reduced in future pneumatic deicer designs.

26. During the PBDS activation cycle at 81 KCAS, engine power increased within 2 seconds to approximately 180 SHP above the steady state value before PBDS activation and resulted in a 4 to 5 rotor rpm transient droop. The power increase equates to approximately 360 SHP more than the standard UH-1H under similar gross weight and density conditions. The excessive power increase during the PBDS activation cycle in forward flight is considered a problem area which should be reduced with future pneumatic deicer designs.

27. During level flight performance testing in the vented condition (failure mode of the PBDS), severe vertical vibrations were encountered at calibrated airspeeds above 92 knots. Testing at higher airspeeds in the vented condition was not attempted due to these vibrations. In the event of PBDS failure resulting in a vented condition, airspeed should be reduced below 80 KCAS and a landing made as soon as practicable.

Autorotational Descent Performance:

28. Autorotational descents were qualitatively evaluated at the conditions presented in table 1 in the vented configuration. The vented boot configuration would occur with an engine failure and subsequent loss of the bleed air pressure needed to provide the vacuum which keeps the boots deflated. After establishing a stable autorotational descent, the PBDS air supply was shut off at the regulator-reliever/shut-off valve to establish the vented configuration which allowed the deicer boot to auto-inflate. Auto-inflation in this configuration results from differential air pressures and centrifugal forces acting on the boot. During the 5 to 10 second period of auto-inflating, the autorotative rotor speed decreased by 4 to 6 rpm, but remained above the lower limit of 294 rpm (collective held full down). Steady state descent rates were approximately 800 to 900 feet per minute greater than those reported in reference 9, appendix A for standard blades. Prevention of deicer boot auto-inflation

(vented condition) in case of engine failure would prevent rpm loss and lower descent rates during autorotation. A means of preventing auto-inflation should be developed prior to operational testing of the PBDS.

Handling Qualities

General:

29. Handling qualities testing was conducted in conjunction with Phase I (structural) testing and Phase II (handling qualities and performance) testing. The handling qualities of the UH-1H helicopter with the PBDS installed were also qualitatively evaluated throughout the dry air testing. Results of these tests were compared with the standard UH-1H data reported in reference 10, appendix A.

Control Positions in Trimmed Forward Flight:

30. Control positions in trimmed ball-centered forward flight were evaluated from 36 to 98 KCAS with the PBDS in the deflated and vented conditions. Tests were conducted in conjunction with the level flight performance at the conditions presented in table 1. Test results are presented in figures 4 and 5, appendix E. The control position characteristics in trimmed forward flight are satisfactory and similar to standard UH-1H helicopters.

Low-Speed Flight Characteristics:

31. The low-speed flight characteristics of the UH-1H helicopter with the PBDS installed were evaluated at the conditions listed in table 1. Tests were performed at a constant skid height of 10 feet in winds of 5 knots or less. A ground pace vehicle with a calibrated fifth wheel was used as a speed reference. Data were recorded at 5-knot increments from a hover to 40 KTAS forward and 20 KTAS sideward and rearward. The low-speed flight data were obtained with the PBDS in the deflated and vented conditions and is presented in figures 6 through 9, appendix E. Variations in the control positions during low-speed flight as a result of venting the boot are essentially the same as the PBDS deflated data and will not be addressed separately. Results are consistent with previously reported data on a standard UH-1H (ref 10, app A) and are satisfactory.

Autorotational Handling Qualities:

32. The autorotational handling qualities were qualitatively evaluated during entry, steady state descent and during simulated

autorotational landings which were terminated with a power recovery. These tests were accomplished with the PBDS in the vented condition simulating an engine failure at entry airspeeds of 70, 80, and 90 knots calibrated airspeed (KCAS). The standard deceleration and pitch application used for autorotational landing practice in a standard UH-1H were effective in slowing the forward ground speed and rate of descent of the UH-1H with the PBDS installed. Rotor speed remained in the normal operating range except with pitch application during recovery and there was no tendency to overspeed during the deceleration. The autorotational handling qualities of the UH-1H with the PBDS installed are similar to standard UH-1H helicopter during steady state descent and simulated autorotational landings to a power recovery, however, increased descent rates will result in modification of the height velocity and glide distance charts for safe engine out operation of the test UH-1H.

Helicopter Response to PBDS Activation:

33. The response of the UH-1H during activation of the PBDS cycle was qualitatively evaluated during hover, low-speed flight and forward flight at airspeeds throughout the established flight envelope. Control fixed aircraft response was evaluated to determine the change in aircraft rates and attitudes under various flight conditions and pilot in the control loop controlling aircraft rate and attitude response was evaluated under similar conditions to determine the level of pilot effort required to control the aircraft during the PBDS activation cycle. Representative time histories of controls-fixed and pilot compensated aircraft response at 81 KCAS forward flight are presented in figures 3 and 10, appendix E, respectively.

34. During a hover the controls fixed aircraft response to PBDS activation generally resulted in a moderate right yaw with rates up to 30 degrees per second. Very little roll or pitch attitude change was observed. This yaw rate was controlled three times by the pilot with the application of approximately 1/2 inch left pedal resulting in an initial heading excursion of ±10 degrees. With increasing airspeed, the yaw rate became less. During forward flight at 81 KCAS the helicopter started a slow right yaw (up to 5 degrees per second) followed by a right roll (maximum rate 10 degrees per second) and small pitch attitude change of two to three degrees. With the pilot in the control loop during the PBDS activation cycle, precise control of helicopter attitudes required inputs to all controls. At 81 KCAS in level flight (fig. 10, app E) approximately 3/4 inch left pedal input was required to initially maintain constant heading control (+5 degrees). As power required reduced to base line over 45 seconds during boot deflation left pedal settings were also

reduced to maintain trim condition. One-half inch lateral cyclic inputs were required to maintain a constant roll attitude (±3 degrees angle of bank) and one-fourth inch longitudinal cyclic inputs were required to maintain a relatively constant pitch attitude and airspeed control (±5 knots). These inputs required during activation of the PBDS in forward flight required moderate pilot compensation (HQRS 4). The aircraft response to PBDS activation during hover and forward flight is a problem area. Aircraft response to PBDS activation during hover and forward flight should be improved in future PBDS designs.

35. Vibration assessments using the VRS (fig. 2, app D) were qualitatively obtained at 10 knot increments from 30 to 100 KCAS in level flight as presented in table 2. Flight at airspeeds greater than 90 KCAS for extended time should not be conducted with the deicers in the deflated configuration due to moderate vertical vibrations encountered above 90 KCAS (VRS 6). With the deicers in the vented configuration airspeed should be kept below 80 knots and landing made as soon as possible. Activation of the PBDS at 90 KCAS and above resulted in severe vertical vibrations (VRS 7 and 8) for approximately 8 seconds during inflation and the start of boot deflation. The PBDS should not be activated at airspeeds of 90 KCAS and greater due to severe vertical vibrations encountered at these airspeeds with the PBDS activated. The vertical vibration levels experienced during activation and also venting of the PBDS is a problem area which should be decreased in future pneumatic deicer designs.

Table 2. Vibration Rating Assessment in Level Flight

Deicer Configuration	Knots Calibrated Airspeed 30	40	50	60	70	80	90	100
Deflated (normal)	2	2	2	2	2	2	3	4
Vented[1]	3	3	3	3	5	6	7	
Inflated[2]	3	3	3	3	5	5 or 6	7 or 8	

NOTES:

[1]Vented condition would result with disruption of bleed air pressure or flow rate.
[2]Deicer boots were inflated during normal PBDS activation.

PHASE III HOVERING ARTIFICIAL ICING TEST

General

36. Inflight artificial icing tests were conducted during January and February 1983 at the NRC icing spray rig in Ottawa, Ontario, Canada to evaluate the ice removal capabilities of the PBDS. The spray rig provided a cloud 75 feet wide by 15 feet high from a nozzle array of 156 steam atomizing water nozzles. As shown in photo 2, the spray rig installation allowed hovering the test aircraft 20 to 30 feet above the ground while exposing the rotor system to the spray cloud when winds were greater than 6 knots. The cloud median volumetric diameter was reported to be 30 microns, and flow rate was adjustable to allow a maximum liquid water content (LWC) of 0.8 gm/m^3. Wind speed and gustiness impact uniformity of the cloud, affect estimation of LWC and changes in wind direction and gustiness move the cloud relative to the hovering aircraft, affecting consistency of rotor immersion. The temperature range of interest for these tests was -5 to -20°C, with LWC values as high as 0.75 gm/m^3.

37. In determining a suitable test procedure, an ice thickness of 0.25 inch at the rotor blade mid-span (blade station 144, 12 feet from the hub) was selected as a baseline condition. Previous UH-1H test data of immersion times in the Ottawa spray rig for this ice thickness were already available in reference 16, appendix A and 0.25 inch accretion at mid-span was known to not seriously affect UH-1H performance. In addition, a 0.25 inch ice accretion is a minimum thickness typically recommended for cycling pneumatic deicing boots on fixed-wing aircraft.

38. To document ice accretions and deicing action of the PBDS inflight, high-speed video and motion picture photography from the ground was used. After shutdown, visual inspection of ice coverage and shapes at mid-span and at the inboard end of the deicer boots (blade station 54) were made. Observations were recorded on data sheets and pertinent photographs were taken.

39. The test procedure involved first immersing the aircraft in the spray cloud for a time interval estimated necessary to accrete a 0.25 inch ice thickness at the rotor blade mid-span point. On examination after shutdown, measured ice thicknesses were not always the desired 0.25 inch and not always available at the mid-span point because of self-shedding and non-uniform cloud immersion. Actual ice accretions varied from 1/16" to 3/8" at mid-span and from 1/64" to 3/16" at the inboard end of the deicer boots. After these initial ice accretions were documented, the aircraft was restarted and a PBDS activation cycle performed to shed the

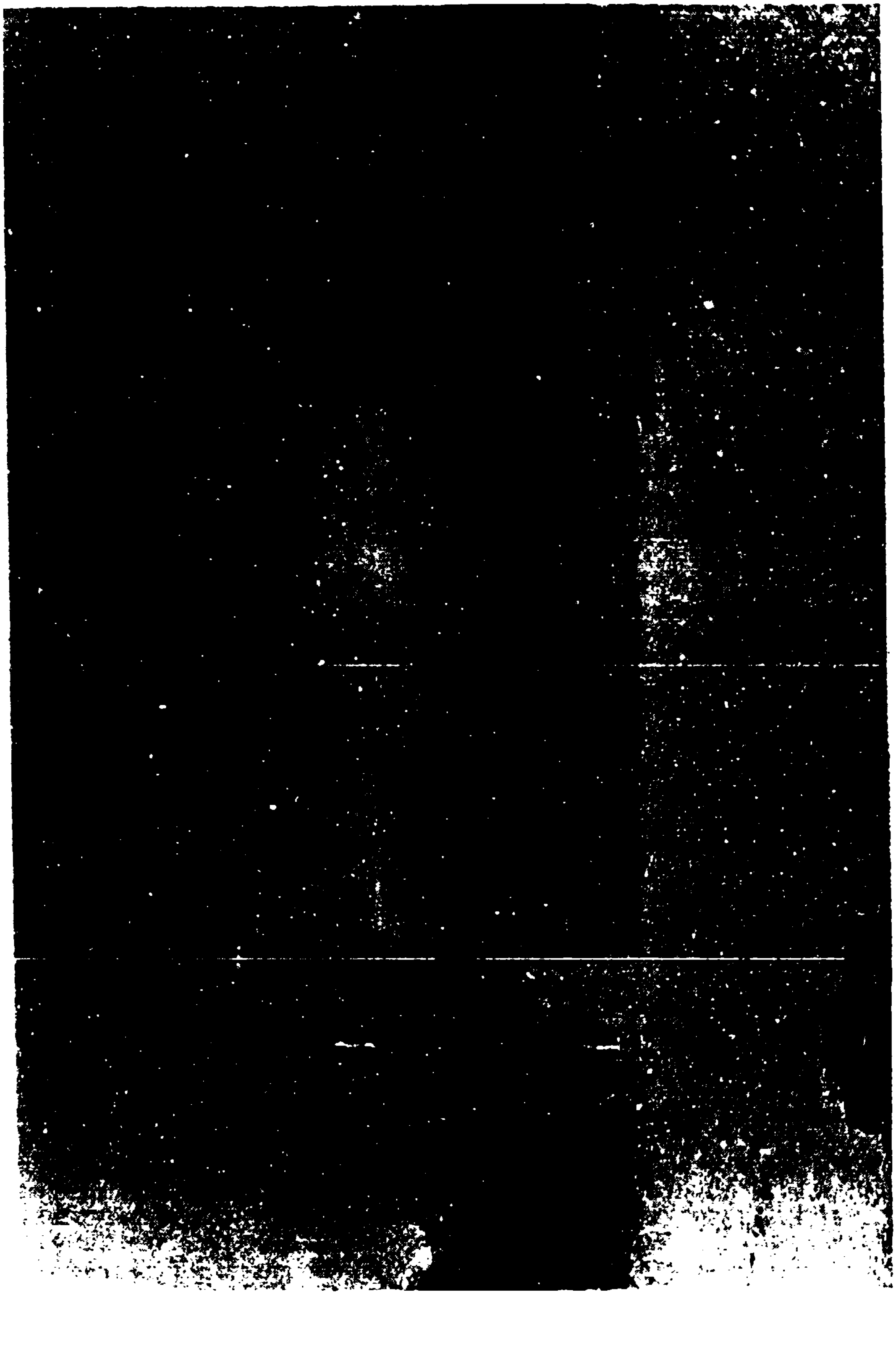


Photo 2. Test Helicopter in Icing Spray Rig Cloud

ice. The aircraft was again shutdown, and an inspection of the ice remaining was made to document the effectiveness of ice removal by the deicer boots. If the blades were not entirely clean, residual ice was removed by inflating the boots with a portable air bottle and then rubbing a leather glove over the deicer boot surface.

40. After this sequence was accomplished, the spray cloud was re-entered for an immersion time necessary to reach either an aircraft operating limit or 15 minutes. The applicable aircraft limits were either an increase of 5 PSI engine torque, an increase in inlet differential pressure of 10 inches of water, or onset of a moderate vibration level resulting from an asymmetric shed. If an aircraft limit was reached or approximately 15 minutes of cloud immersion had elapsed, the aircraft exited the cloud and a PBDS activation cycle was performed. PBDS activation cycles were documented with high-speed video. The aircraft was then shutdown to inspect any ice remaining.

41. An additional evaluation similar to that described above was performed with the deicer boots treated with ICEX®, a silicone base polymer proprietary to BFG. ICEX® is an ice adhesion depressant specifically formulated to bond to the deicer boot surface, and reportedly reduces the adhesive force of the ice on the deicer surface to less than one-sixth that of an untreated surface.

42. The artificial icing tests were performed at the temperature and LWC conditions shown in table 3 and figure D. A total of 39 individual icing cloud immersions and 43 PBDS activation cycles were completed. Overall, the deicing capability of the deicer boots was satisfactory at all conditions tested in the spray rig. Two problem areas were identified during Phase III: (1) breakdown of internal pneumatic deicer ventilation material during blade rotation and (2) loss of regulated PBDS pressure during flight. Loss of regulated PBDS pressure was corrected at the test site during Phase III, however, internal breakdown of the pneumatic deicer will require further investigation by BFG.

43. At accretion levels greater than 0.25 inches at mid-span, activation of the PBDS resulted in complete removal of the ice cap from the leading edge of the main rotor. As shown by high speed photography, ice fracturing began inboard as the pneumatic deicer inflated. At lower accretion levels, ice of less than 0.06 inches was sometimes retained, as shown in photo 3, but subsequent deicer cycles after further ice accretion during multiple immersions removed these accumulations.

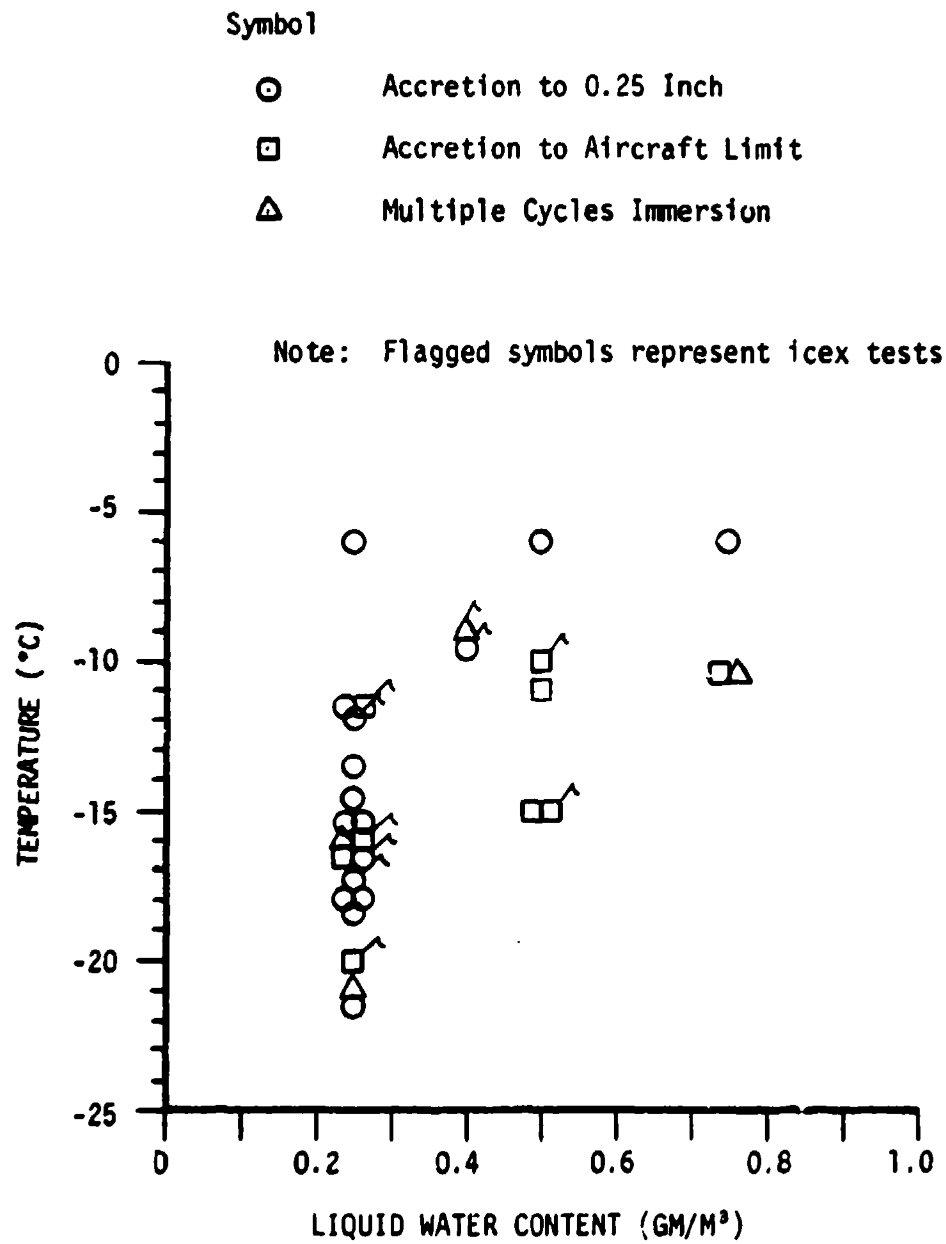
Table 3. Spray Rig Icing Test Conditions

LWC at Aircraft (gm/m^3)	OAT (°C)	Equivalant Immersion Time (min)	Wind Speed and Gustiness[1] (mph)
0.25	-13	4	15 M[1]
	-13.5	5	15 M
	-12.5	5 1/2	10 M
	-12.5	6	10 M
0.25	-17.5	14	10 M
	-16	8 1/4	10 M
	-15.5	3 1/2	10 M
	-15.5	5 1/2	10 M
0.25	-18	10 1/2	10 M
	-18	8 1/2	10 M
	-18.5	6 3/4	10 M
0.25	-21.5	12 1/2	7 L
	-21.5	5 1/2	10 M
	-21	6 1/2	10 M
	-20.5	6 1/2	10 M
	-20	7	10 M
	-16	21 1/2	10 M
	-16.5	10 1/2	10 M
0.25	-6	10 1/2	5 L
0.25	-14.5	3 1/2	5 L
	-13.5	6	5 L
	-11.5	5 1/4	5 L
0.25[2]	-12	5	10 M
	-11.5	15 1/2	10 M
0.25[2]	-16.5	4 1/2	10 M
	-17.5	4 1/2	7 M
0.25[2]	-20	14	5 L
0.40[2]	-9.5	4	11 M
	-9	13	11 M
0.50[2]	-16	15	10 M
	-15	9	7 M
0.50	-6	7 3/4	5 L
0.50[2]	-11	12 1/2	5 L
0.50	-11	16	10 M
0.50	-15	7 3/4	10 M
0.75	-6	12	5 L
0.75	-10.5	9 1/2	8 M
	-10.5	10 1/4	8 M

NOTE:

[1]Gustiness: Low (L) = less than ±1.5 mph.
Medium (M) = ±1.5 to ±3 mph.
[2]ICEX® Tests

Figure D. Spray Rig Icing Test Conditions.



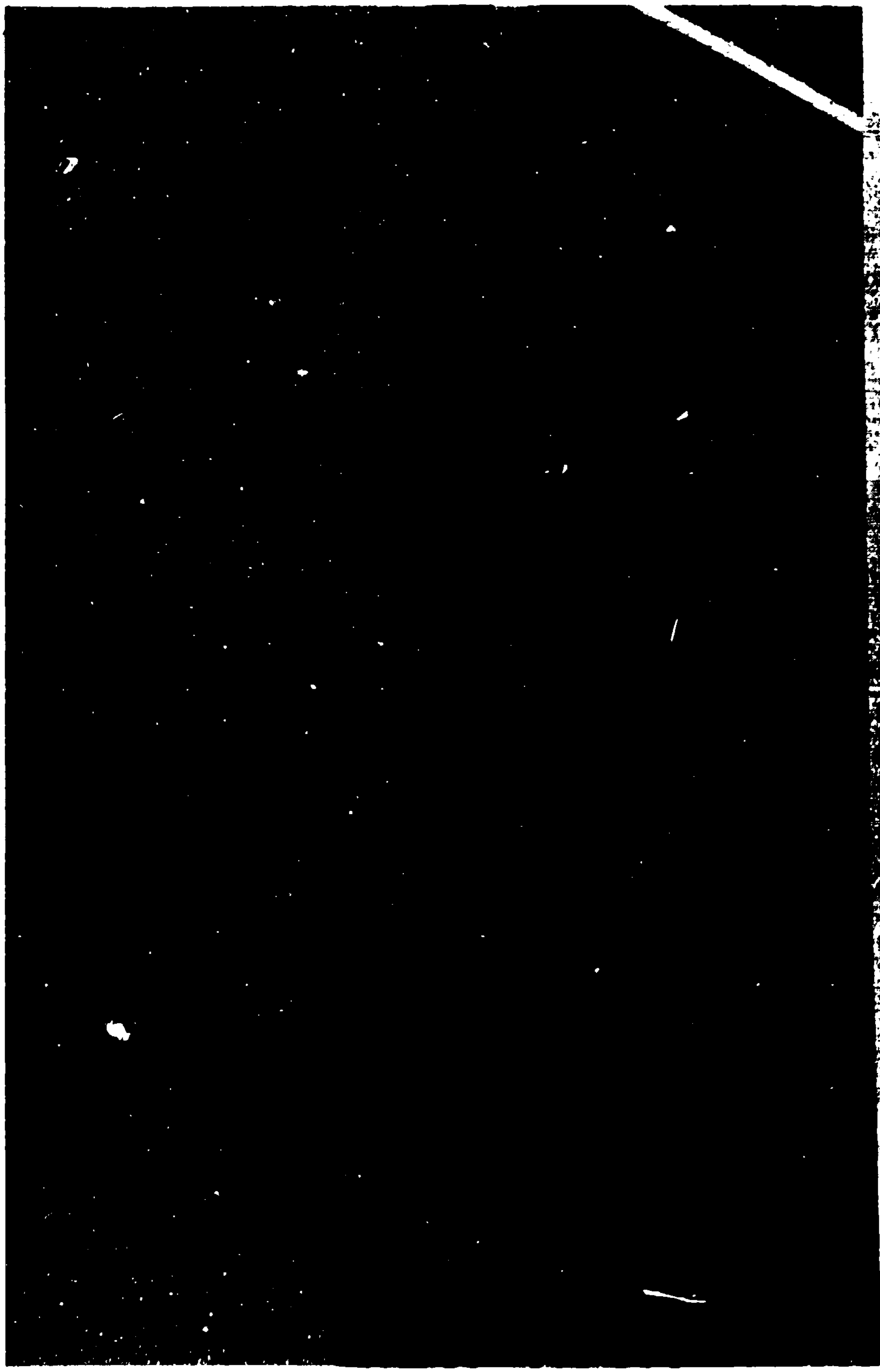


Photo 3. Residual Ice after Deice Cycle

44. The amount and thickness of remaining ice after a single deice cycle at low accretion levels is insignificant, as shown in figures E, and F. High-speed photography revealed that ice fracturing during PBDS activation occurred differently at the lower accretion levels, with ice leaving the deicer boot near blade station 84 first, then inboard and outboard almost simultaneously.

45. During the immersion times intended to reach an aircraft limit, the 5 PSI torque rise caused by ice accretion was never reached due to natural ice sheds and/or ice shapes that conformed to the main rotor blades. All PBDS activations resulted in symmetrical ice sheds without increases in lateral vibrations. Activation of the pneumatic deicer boots after natural asymmetrical ice sheds reduced the associated vibration within two seconds and is an enhancing characteristic. Aircraft power required increases experienced during deice cycles were the same as seen during hover performance.

46. Limited flight testing was conducted at -5°C, however, natural ice shedding occurred frequently enough that PBDS activation was not necessary. All single deice cycles resulted in clean blades without residual ice remaining on the deicer boot. Weather at the test site allowed for one afternoon of icing tests at the -5°C condition, permitting a total of only 0.5 hour of flight time in the spray cloud at -5°C; therefore, multiple cycles immersion and ICEX® testing were not completed at this temperature.

47. At -20°C and 0.25 gm/m^3 LWC, ice accreted to almost full span on the main and tail rotor blades as shown in photos 4 and 5. With higher temperatures the spanwise accretion is reduced. At -10°C ice accreted to approximately 70% to 80% of the main rotor blade span and 50% to 60% at -5°C in the hover condition.

48. The outboard section (past blade station 192) may be self-shedding prior to accreting 0.25 inches at the midspan during icing tests. If the outboard blade area self-sheds continuously and symmetrically from the ESTANE® surface, then a spanwise reduction of the pneumatics within the ESTANE® boot may be in order with future design changes. A spanwise reduction of the pneumatics, especially in the higher lift and velocity areas of the main rotor blade, would probably reduce profile drag, vibration levels, ESTANE® erosion, deflation times, and improve handling qualities during PBDS activation. An investigation should be made to determine the feasibility of reducing the spanwise distribution of the pneumatic deicer boots in future pneumatic deicer changes.

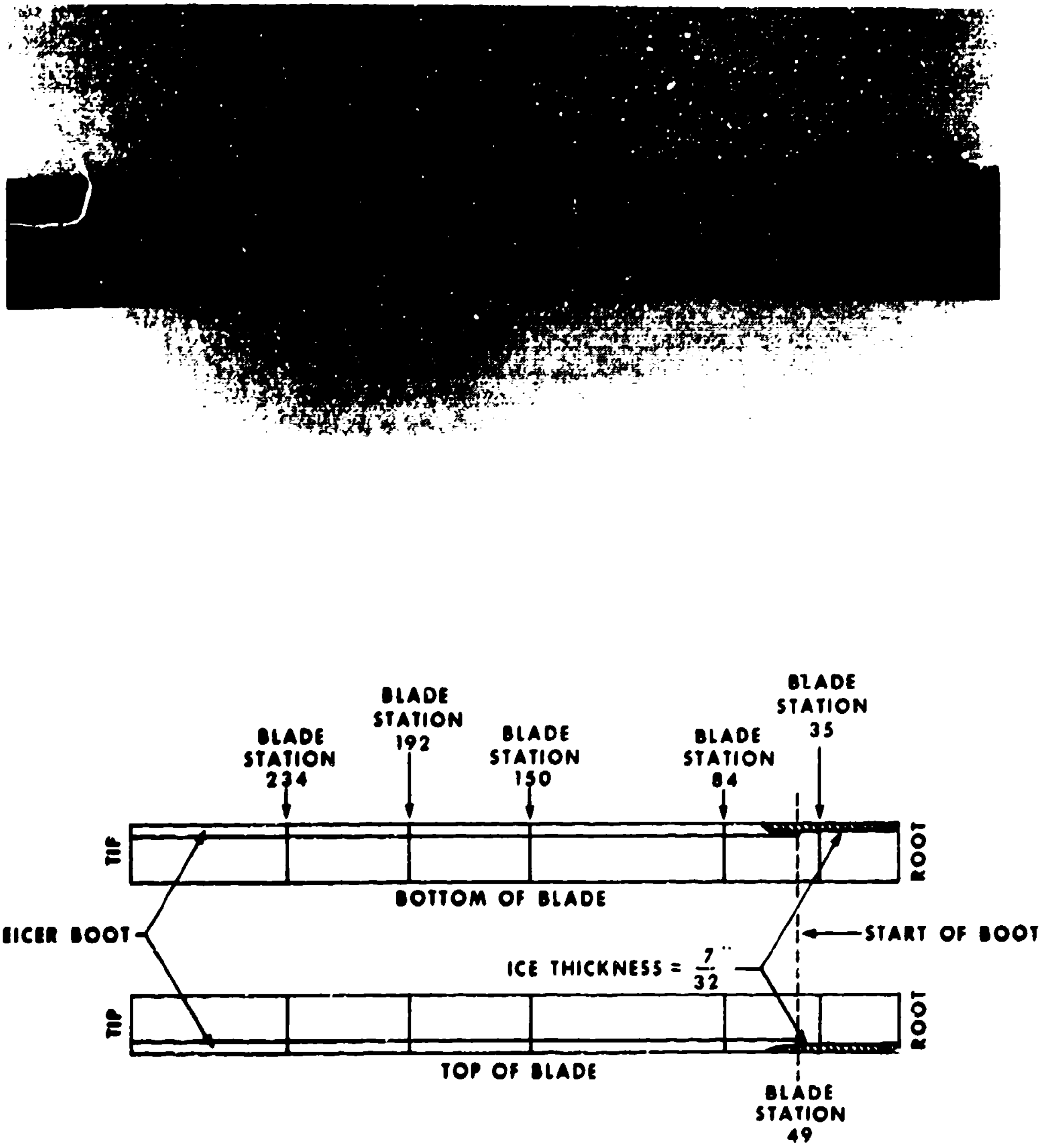


Figure E. Residual Ice after Deice Cycle at -12°C with Clean Blades

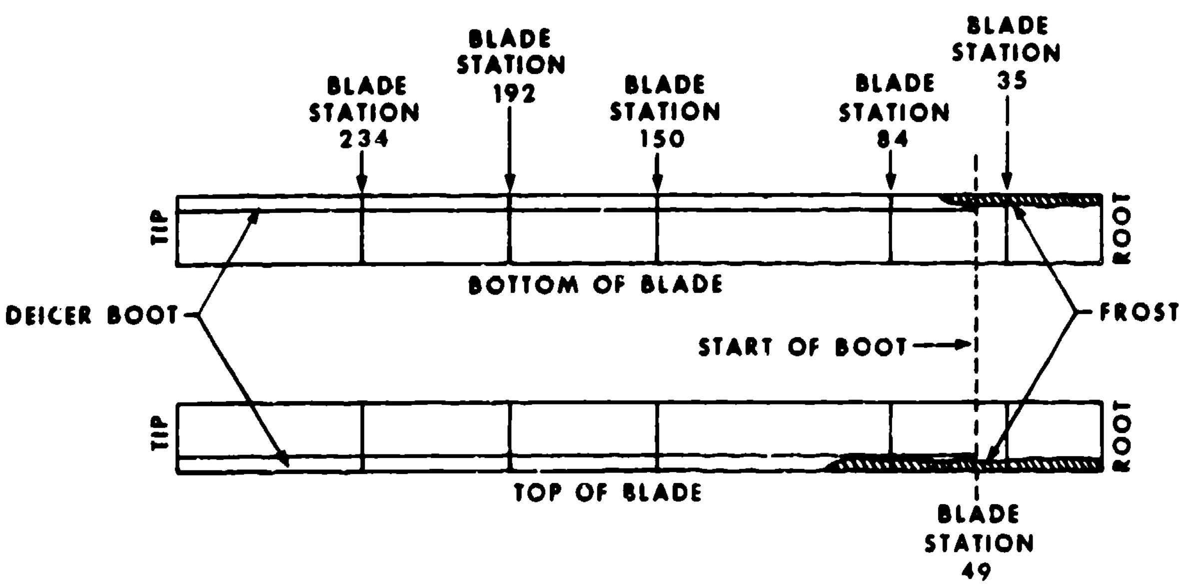


Figure F. Residual Ice after Deice Cycle at -15°C with ICEX Applied to Blades

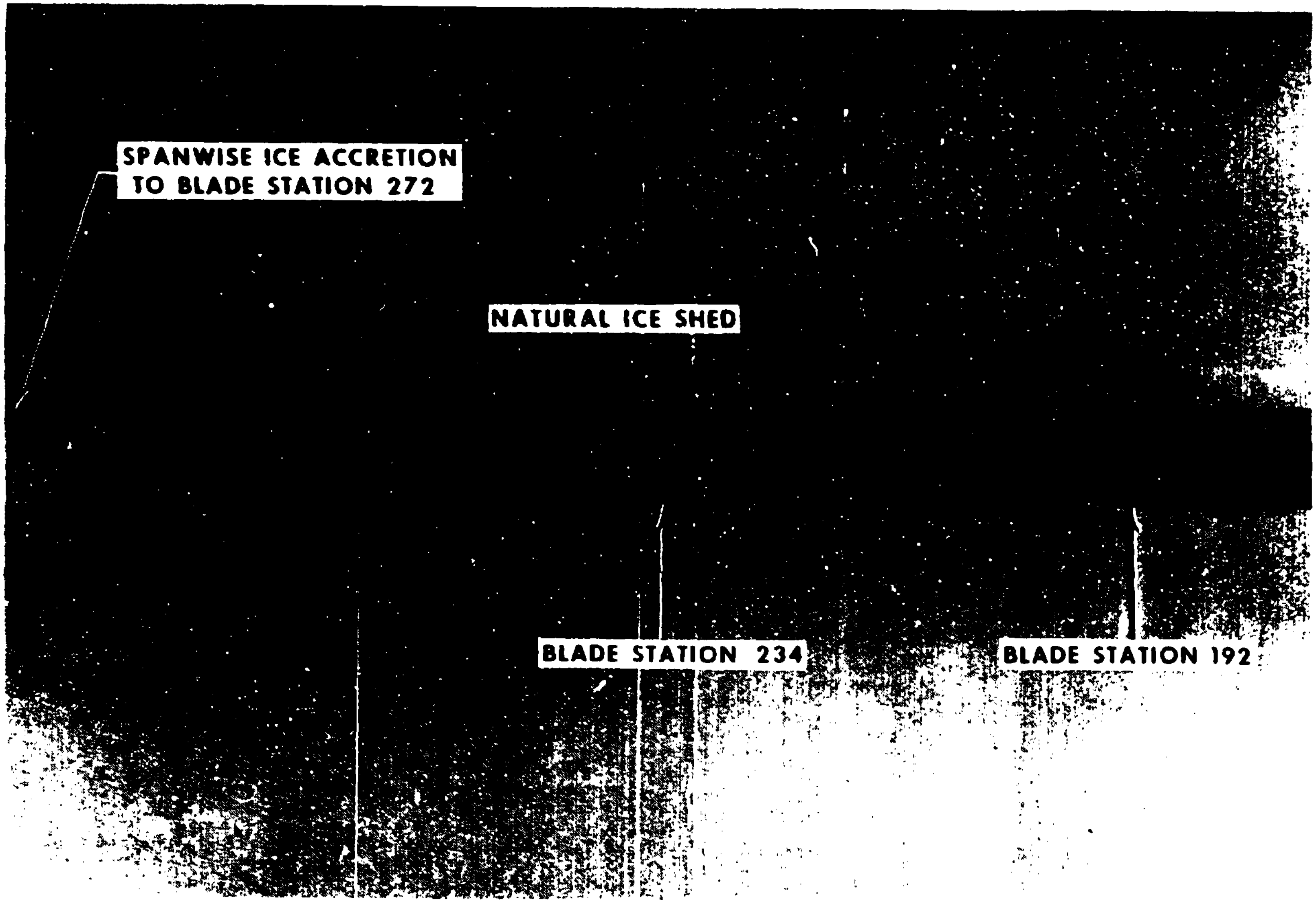


Photo 4. Ice Accretion and Natural Shed Areas at −20°C

Photo 5. Ice Accretion on Tail Rotor at -20°C

ICEX® TESTING

49. ICEX® appeared to promote earlier ice shedding and reduction of residual ice after deicing, and lengthened the elapsed time to start of moderate vibration levels. The erosion and flow characteristics of the ICEX® compound were documented after each flight. A strip of 2 inch tape (3M No. 355) was pressed to the rotor blade surface to qualitatively evaluate its adhesive strength. If the tape adhered to the blade surface similar to a clean dry blade surface, then the ICEX® was considered to be eroded. ICEX® was re-applied to the full span of the deicer boot daily. ICEX® erosion appears to be similar to that found during ice phobic testing reported in reference 17, appendix A. After a total immersion time of 36 minutes at 0.25 and 0.50 gm/m^3 in the icing cloud at -11°C with light snow, the ICEX® was eroded from the leading edge of the rotor blade from the tip to blade station 168 as shown in figure G. ICEX® may be useful in reducing the small amount of residual ice found on the pneumatic deicer after a deice cycle found in the lower g and velocity areas from blade station 84 inboard. The susceptibility of ICEX® to erosion on the UH-1H main rotor blade is a problem area and should be corrected prior to use of ICEX® in the icing environment.

50. While conducting tests at the spray rig, a decrease in system pressure to 13 PSI was experienced during a PBDS activation. The reduced pressure output diminished the deicer effectiveness, leaving approximately a 36 inch length of 0.20 inch thick ice on either side of the 50% span point on the main rotor blade. The PBDS regulator had malfunctioned internally allowing bleed air to vent overboard. Loss of PBDS pressure due to a malfunctioning regulator is discussed further in the reliability and maintainability section of this report.

RELIABILITY AND MAINTAINABILITY

General

51. The reliability and maintainability of the PBDS were evaluated during all testing phases. Phase I testing was performed using three successive modifications (app B) to the internal deicer design which were intended to reduce the deicer deflation time interval. Phase II included testing with modifications three and four while Phase III was conducted entirely with the fourth deicer modification.

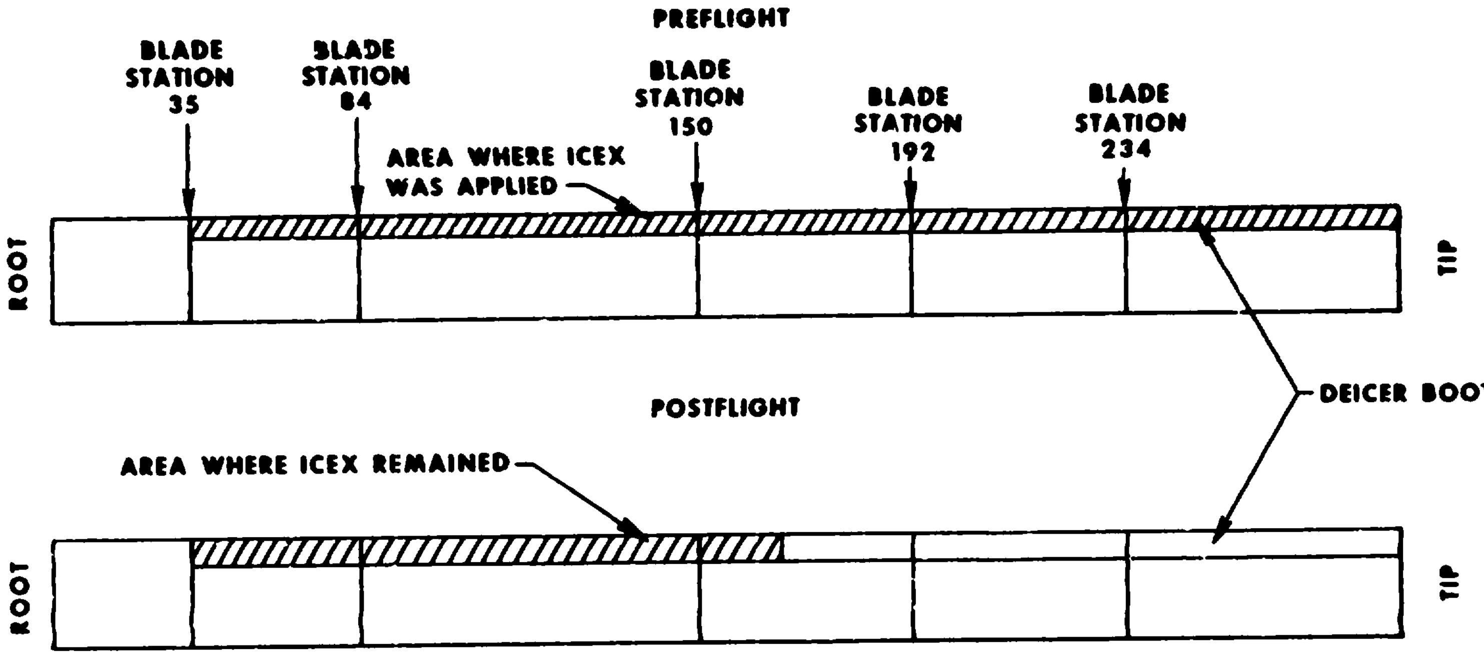


Figure G. ICEX Erosion

Phase I Testing:

52. During hover and low airspeed structural tests the leading edge of the pneumatic deicer boot beyond blade station 240 exhibited erosion and small punctures of the ESTANE® material creating a loss in system vacuum and pressure. Deicer erosion primarily occurred during structural survey testing involving flight over hard surface areas at EAFB and involved sustained inflation of the pneumatic deicer close to ground level. Sustained inflation of the deicer near ground level may increase the susceptibility of the pneumatic deicer to erosion and would not be a normal operating environment or mode for the PBDS. While erosion continued to a lesser degree during Phase II and Phase III testing, sufficient testing time was not available to properly address deicer erosion due to limited test scope and repeated modifications requiring removal of the deicer boot from the main rotor blade. These small punctures were readily repaired by BFG using cold or hot patches, however, the susceptibility of the boots to erosion damage is a problem area. Further investigation of the erosion properties of the pneumatic boot in both the sand and rain environments should be accomplished as soon as possible.

53. During a PBDS operational check of a modified ejector control valve, one pneumatic deicer boot ruptured (blow out) as shown in photo 6. This created a slight rotor vibration with loss of system vacuum for deicer deflation. This occurred on a sunny day with outside air temperature (OAT) of +25°C and winds of less than 5 knots. The flexible ESTANE® material appeared to have separated from the fabric backing and formed into a 0.75 inch bubble that burst upon inflation. BFG engineers speculated that the black deicer surface may have become heated above 65.6°C (critical bonding temperature of only 65.5°C could be a problem in itself) while the aircraft was sitting on the ramp prior to run-up, causing the deicer surface to debond and rupture. BFG replaced the defective pneumatic boot and recommended that the pneumatic boot no longer be activated at outside air temperatures greater than +15°C. The rupture of the pneumatic boot is a problem area. Further investigation to determine environmental limitations regarding pneumatic deicer surface temperatures should be conducted prior to release for field operations.

54. Two of the deicer modifications as discussed in paragraph 15 provided a limited opportunity to assess the retrofit potential of the pneumatic deicer. Eight man-hours were spent preparing the main rotor blade surface which included removing old paint (new blade) with Methyl Ethyl Ketone, sanding the blade surface with 250 wet and dry sandpaper and then wiping the blade with toluene. Two coats of a 3M adhesive (3M 1300L) were then applied to the leading edge of the blade surface and backing of the

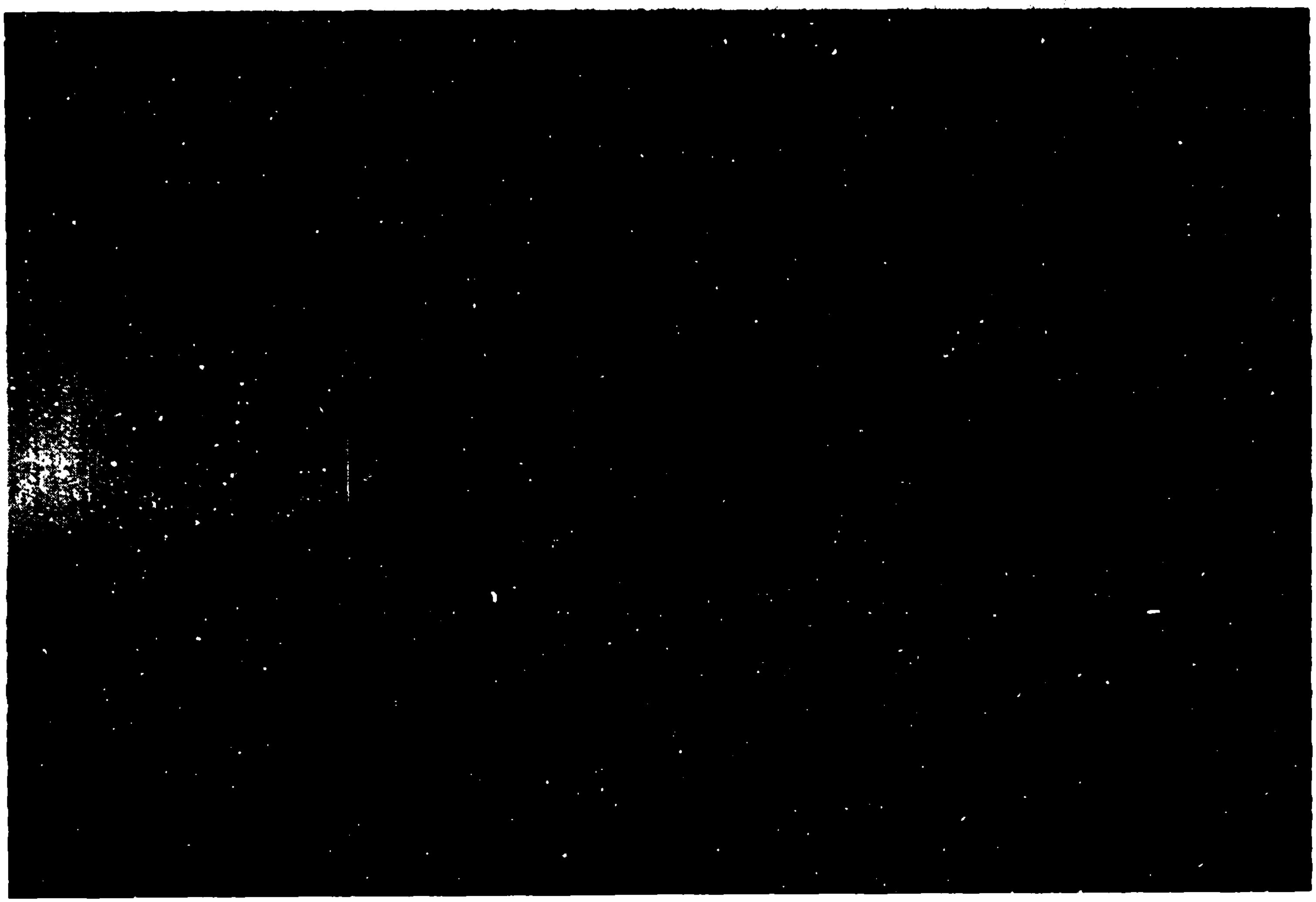


Photo 6. Deicer Boot Rupture

deicer boot. The blades were allowed to set overnight while the adhesive dried to complete the blade preparation. Eight man-hours were then spent placing the deicer boot on the main rotor blades after activating the 3M adhesive with toluene. BFG recommends 48 additional hours prior to inflation of the deicer for inspection and use.

55. During flight and prior to activation of the PBDS the system pressure was manually adjusted to 25 PSI at the regulator. This process was extremely awkward for the pilot or copilot to accomplish so an extra crewmember was required to monitor/adjust PBDS pressure during flight. The requirement to constantly monitor/adjust PBDS pressure during flight is a problem area. Future designs should incorporate automatic system pressure regulations that will compensate for engine bleed air pressure changes which affect system pressure.

Phase III Testing:

56. While conducting artificial icing tests pressure delivered to the pneumatic deicer decreased to 13 PSI during a PBDS activation. The reduced pressure output diminished the deicer effectiveness leaving approximately a 36 inch length of 0.20 inch thick ice on either side of the 50% span point on the main rotor blade. The diaphragm within the PBDS pressure regulator became displaced to one side of the regulator allowing bleed air to port out the pressure relief vent shown in figure 1, appendix B. The diaphragm was easily repositioned within the regulator and made serviceable for continued icing tests. Loss of PBDS pressure to the pneumatic deicers due to a malfunctioning regulator should be further investigated and corrected prior to operation in the natural icing environment.

57. During Phase III testing, lumps developed in the deicer boots near the blade tips. These lumps were caused by a breakdown of the internal ventilation material. Pieces of this material were being forced to the blade tips by centrifugal forces. The lumps were reduced in size by BFG personnel cutting into the deicer boot, applying shop air to keep the boot inflated, digging the pieces of material out of the boot, and then patching the boot. This breakdown of the internal ventilation material is a problem area. Breakdown of internal ventilation material will require correction prior to further testing.

CONCLUSIONS

GENERAL

58. The following conclusions were reached upon completion of the UH-1H PBDS testing.

a. The concept of operation of the PBDS as a deice system is feasible under the conditions tested when installed on the test UH-1H helicopter.

b. Thirteen problem areas were identified of which only two were corrected during the evaluation.

PROBLEM AREAS

59. The following problem areas associated with the PBDS installation were identified, but not corrected.

a. breakdown of the internal pneumatic deicer ventilation material (para 42)

b. erosion of the ESTANE® material from the outboard leading edge of the UH-1H main rotor blade (para 20)

c. loss of regulated PBDS pressure during flight (para 56)

d. the increase in power required for level flight with the PBDS installed (para 25)

e. the large power increase during PBDS activation cycles in forward flight (para 26)

f. the increase in power required to hover with the PBDS installed (para 23)

g. the vertical vibration levels experienced during activation of the PBDS in forward flight (para 35)

h. the aircraft response to PBDS activation during hover and forward flight (para 34)

i. excessive pneumatic deicer deflation time (para 14)

j. reduction of system pneumatic pressure (para 16)

k. the requirement to constantly monitor/adjust PBDS pressure during flight (para 55)

RECOMMENDATIONS

60. The following recommendations are made:

a. Correct the internal breakdown of the pneumatic deicer by BFG prior to further testing (para 57).

b. Investigate erosion properties of the pneumatic boot in sand and rain environments as soon as possible (para 52).

c. Improve reliability of pressure regulator prior to operational testing (para 56).

d. The increase in power required for level flight with the PBDS installed should be reduced if the pneumatic deicer design is to be pursued further (para 25).

e. The power required increase during PBDS activation cycle in forward flight should be reduced if the pneumatic deicer design is to be pursued further (para 26).

f. The power required to hover with PBDS installed should be reduced if the pneumatic deicer deeing is to be pursed further (para 23).

g. Vertical vibration levels experienced during activation and also venting of the PBDS should be decreased (para 35).

h. The aircraft response to PBDS activation during hover and foward flight should be improved (para 34).

i. The deicer deflation time should be reduced prior to future icing tests (para 14).

j. The modification of the ejector control valve should be accomplished to achieve the nominal system pressure of 25 PSI after heat soak prior to icing tests (para 16).

k. Incorporate automatic system pressure regulators that will compensate for engine bleed air pressure changes (para 55).

l. A means of preventing deicer boot auto-inflation should be devised prior to operational testing of the PBDS (para 28).

m. The airspeed should be reduced below 80 KCAS and a landing made as soon as practicable if PBDS fails in vented condition (para 27).

n. Investigate reducing the spanwise coverage of the pneumatic deicer boots (para 48).

o. Investigate deicer surface temperature operational limitations prior to operational testing (para 53).

p. The susceptibility of ICEX® to erosion be reduced prior to operation in the icing environment (para 49).

APPENDIX A. REFERENCES

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APPENDIX B. DESCRIPTION

GENERAL

1. The test helicopter, US Army S/N 70-16318, was a production UH-1H modified to accommodate test instrumentation, the Pneumatic Boot Deicing System (PBDS) installation, rotor brake and a partial ice protection system (Kit A). The principal modifications included the PBDS components installed within the cabin area, routing of bleed air through the mast and rotor assembly, and the pneumatic deicers installed on the main rotor blades. Photos 1 and 2 show the test aircraft with the PBDS and test instrumentation installed.

PNEUMATIC BOOT DEICING SYSTEM

2. The PBDS installation consisted of six major components: the PBDS regulator-reliever/shut-off valve (fig. 1), ejector flow control valve (fig. 2), timer (fig. 3), rotary union (fig. 4), hose and flap assembly (fig. 5) and the pneumatic deicer (figs. 6 through 8). Schematics of the pneumatic deicer system layout designed for evaluation purposes are presented in figure 9.

3. Customer bleed air from the aircraft engine is routed through a check valve to the regulator-reliever/shut-off valve, which is manually adjusted to the system operating pressure of 25 PSI. The solenoid-operated ejector flow control valve (electrically energized by the timer) ports bleed air through the pneumatic rotary union to the deicers in a single inflation activation cycle. A normal activation cycle consists of inflation of the deicer boots for approximately two seconds, followed by subsequent deflation. With the solenoid de-energized, the ejector flow control valve provides the vacuum necessary to keep the deicers deflated by porting bleed air overboard. A shop air test connection is installed downstream of the check valve for leak and maintenance checks. An electrical/manual gate valve is provided upstream of the rotary union to capture pressure or vacuum in the deicer boots and prevent deflation during performance testing or leak checks. The pressure gauge downstream of the regulator-reliever/ shut-off valve displays regulated pressure of the ejector control valve. A vacuum/pressure gauge downstream of the gate valve displays the vacuum or pressure of the deicer boots. To evaluate a degraded mode that would occur with engine failure or damage to the PBDS, vacuum to the deicer boots could be removed by use of the regulator-reliever/shut-off valve. This allows the deicer boots to vent to ambient atmospheric conditions through the de-energized ejector control valve, resulting in partial boot inflation (auto-inflation) from differential air pressures and centrifugal forces on the rotor blades.

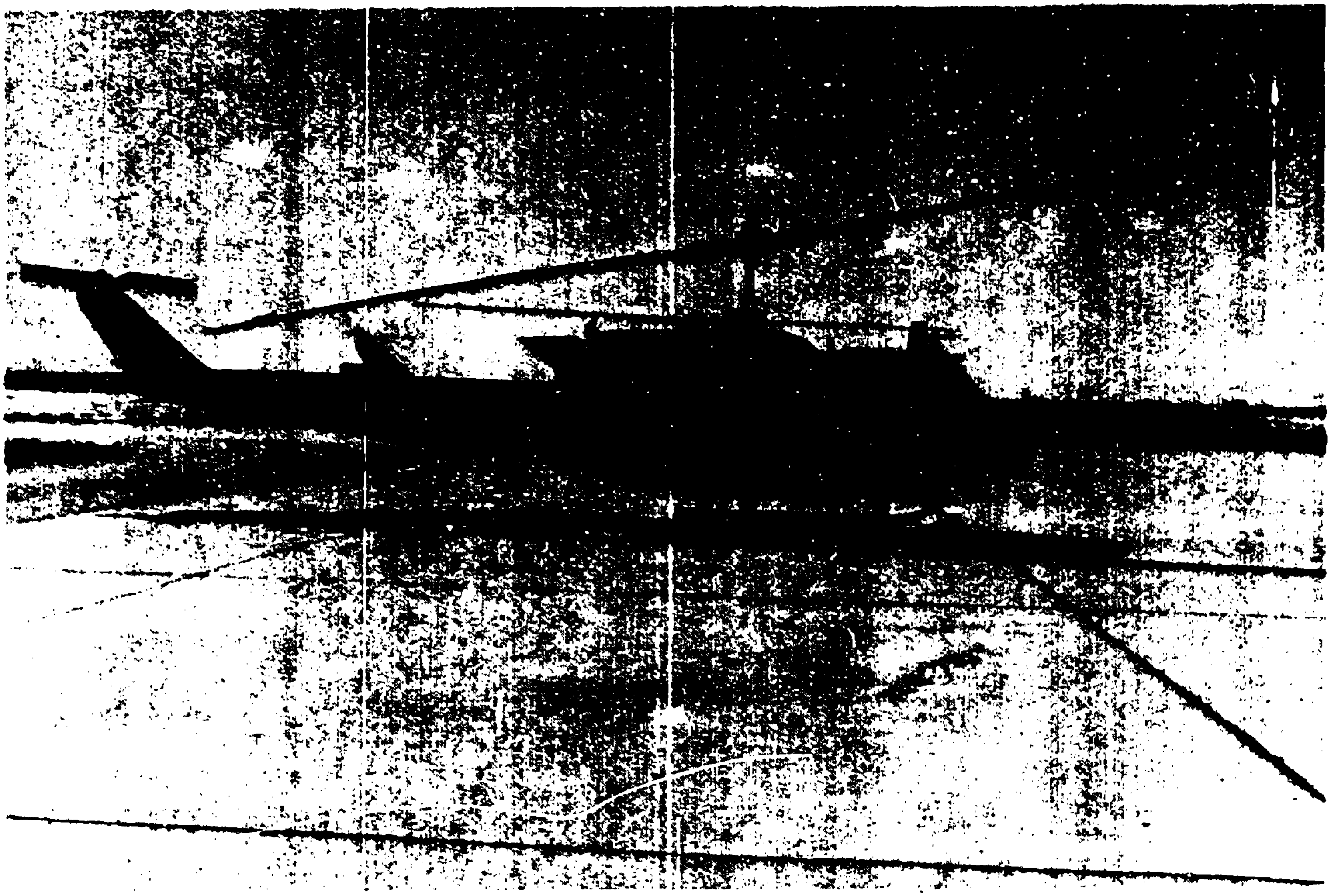


Photo 1. Test Aircraft with the PBDS and Test Instrumentation Installed (Side View)

Photo 2. Test Aircraft with the PBDS and Test Instrumentation Installed (Front View)

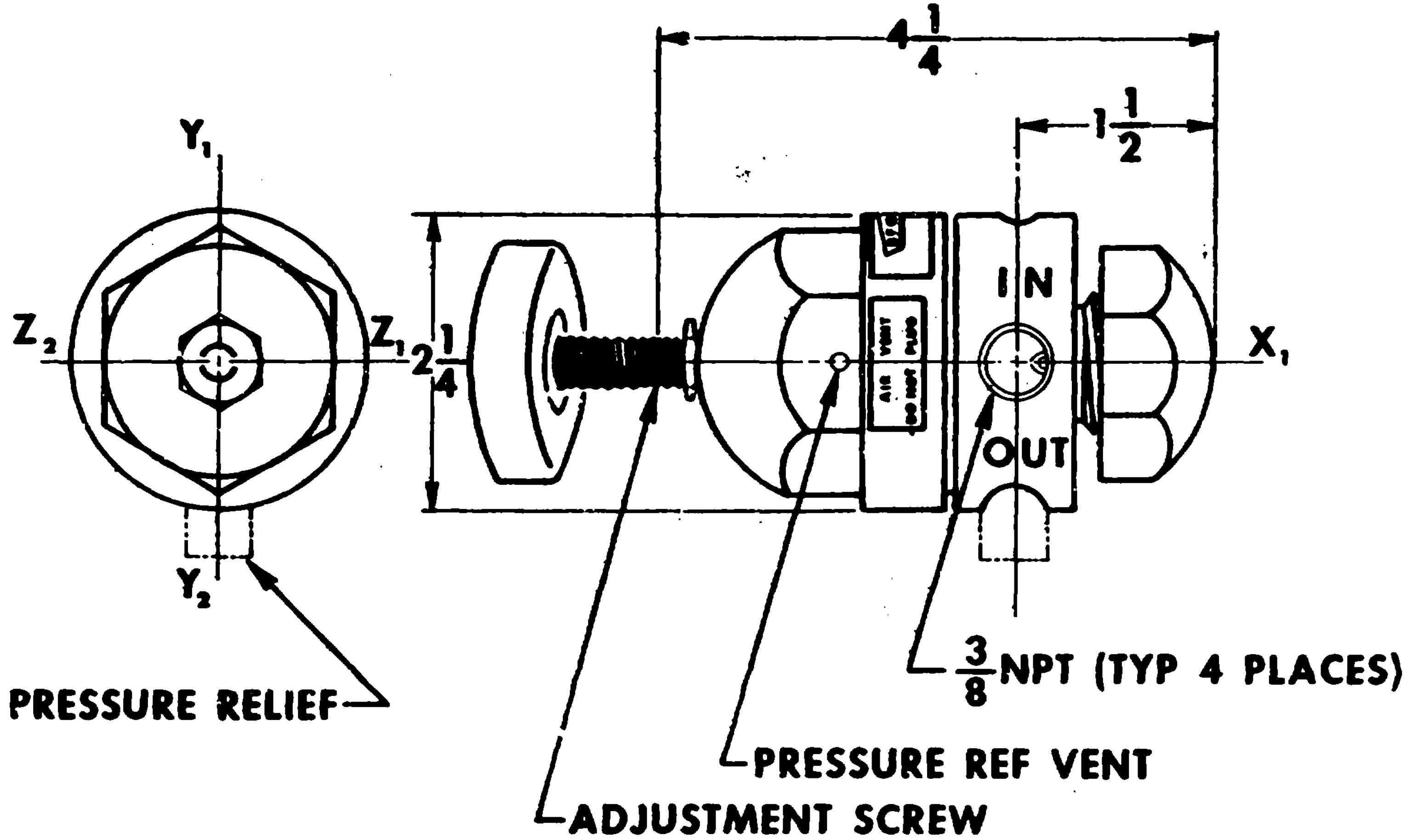


Figure 1. Regulator-Reliever/Shutoff Valve

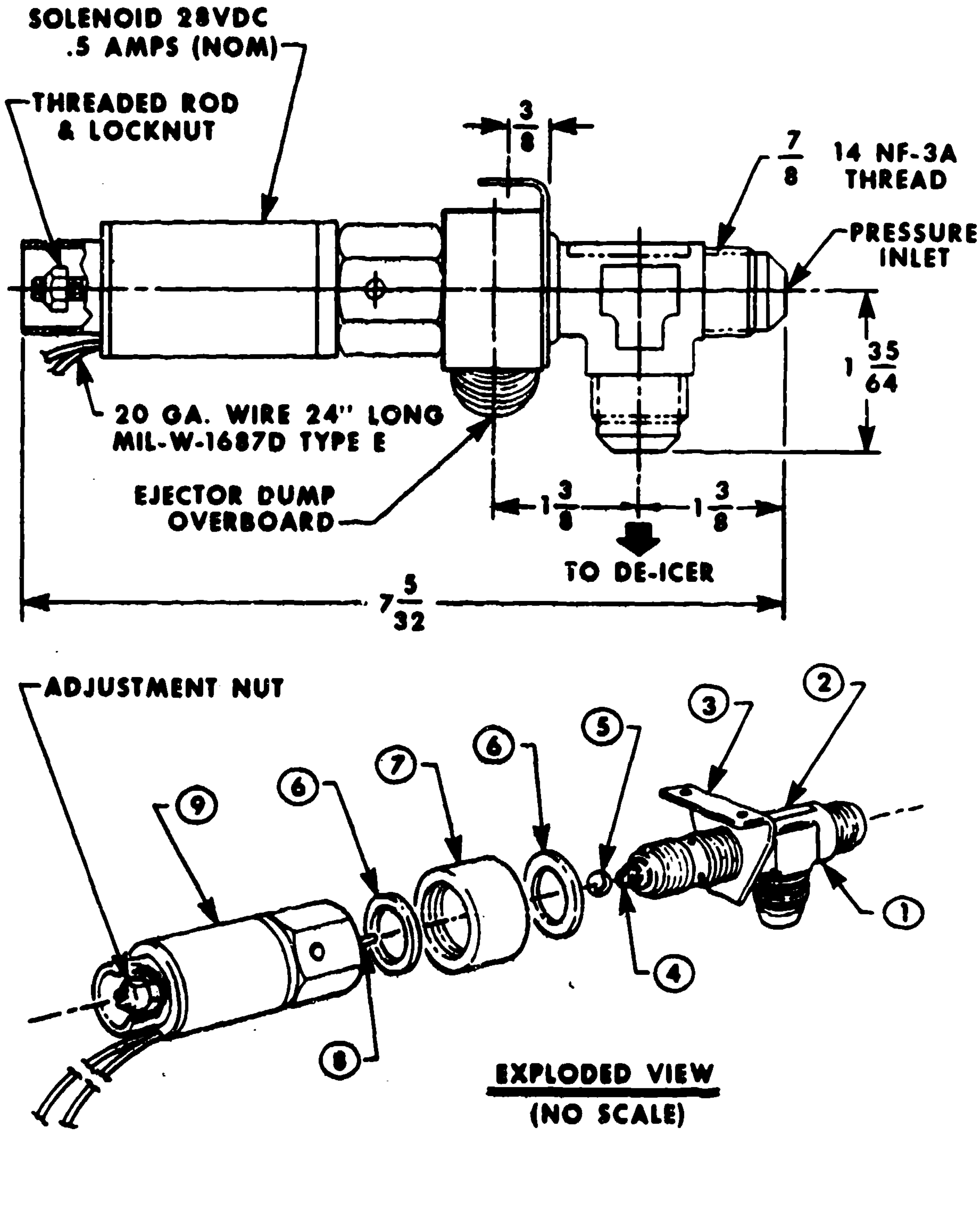


Figure 2. Ejector Flow Control Valve

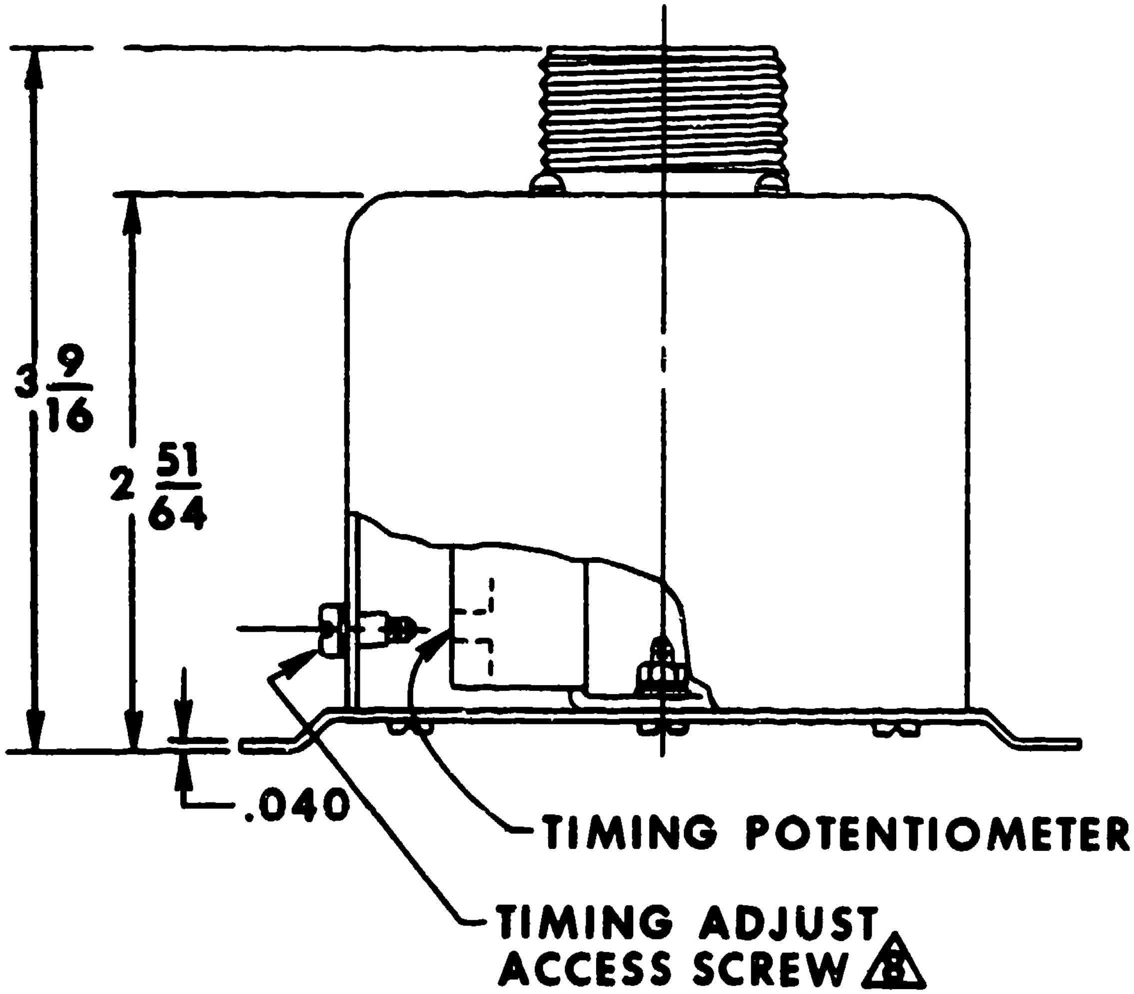


Figure 3. PBD8 Timer

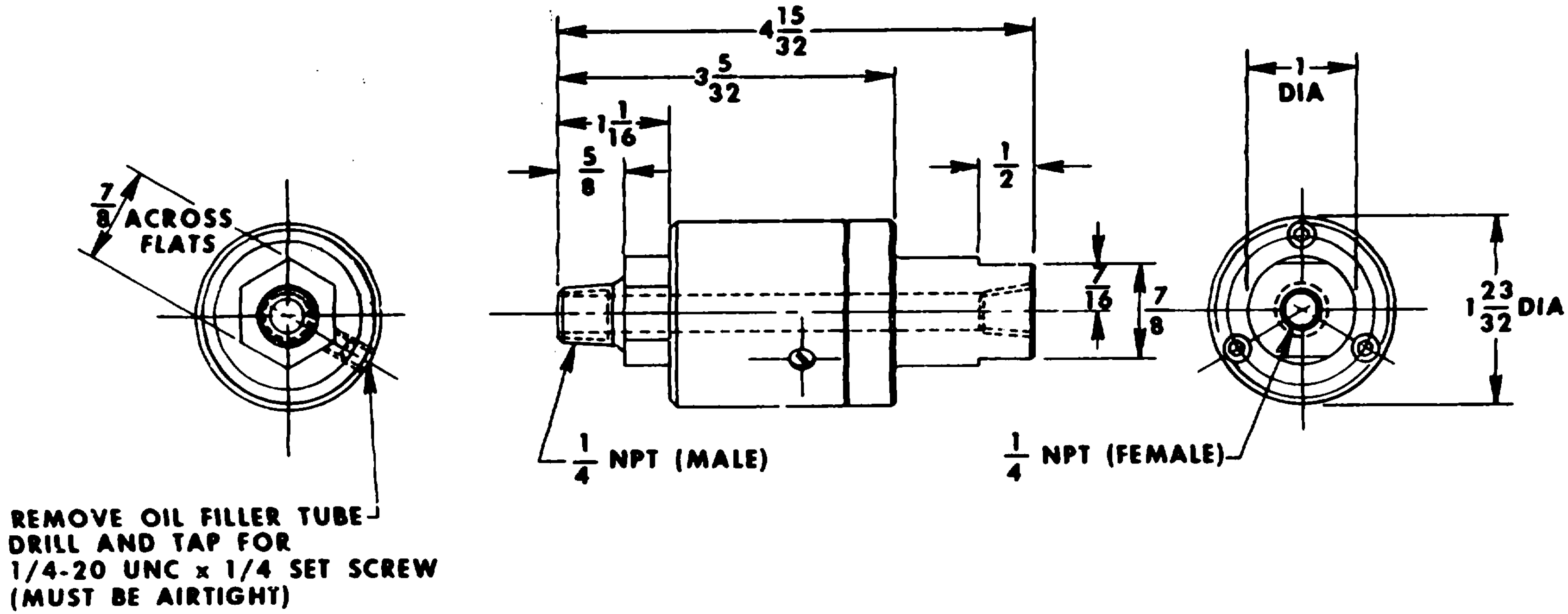


Figure 4. Rotary Union

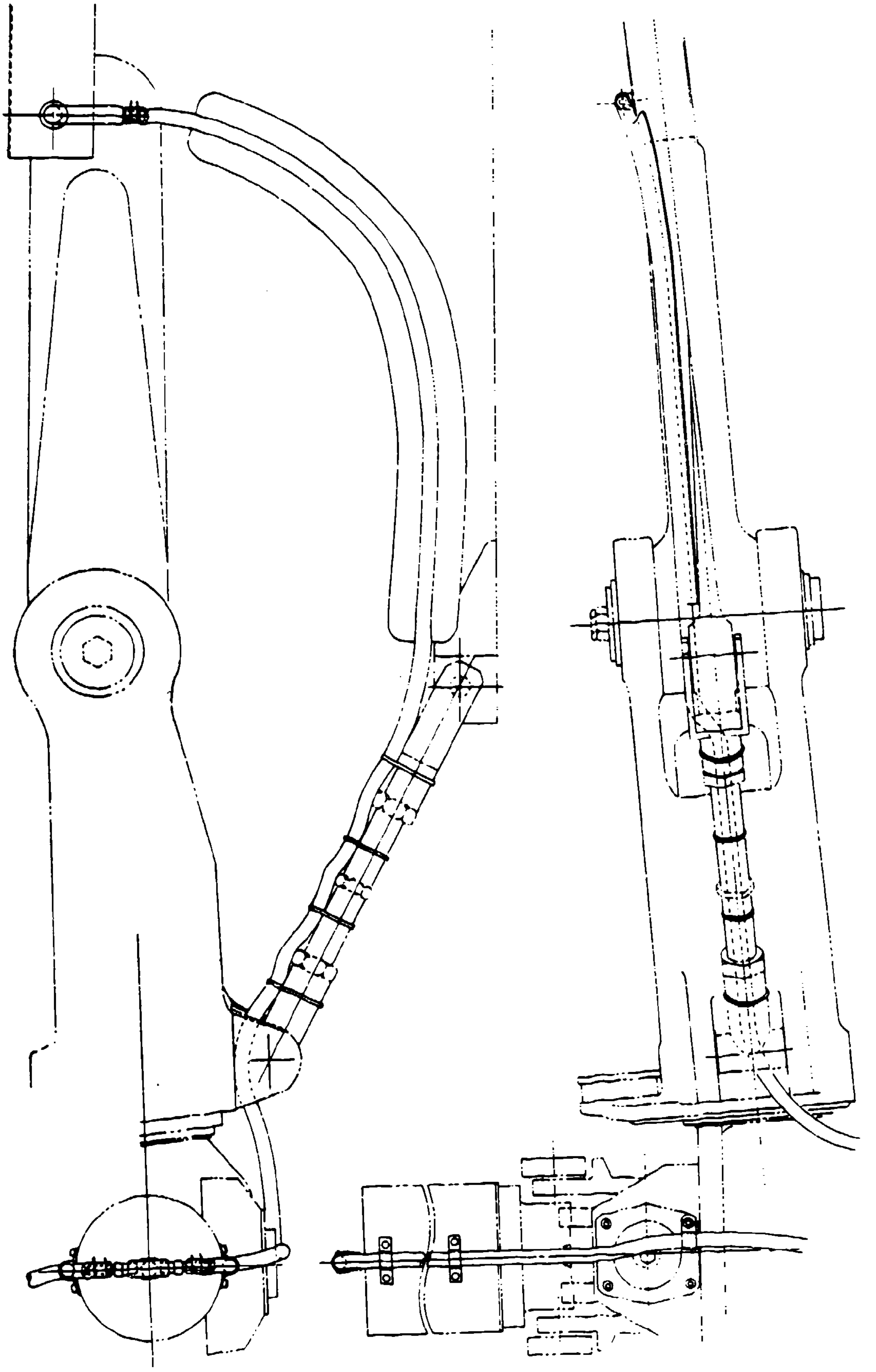


Figure 5. Hose and Flap Assembly

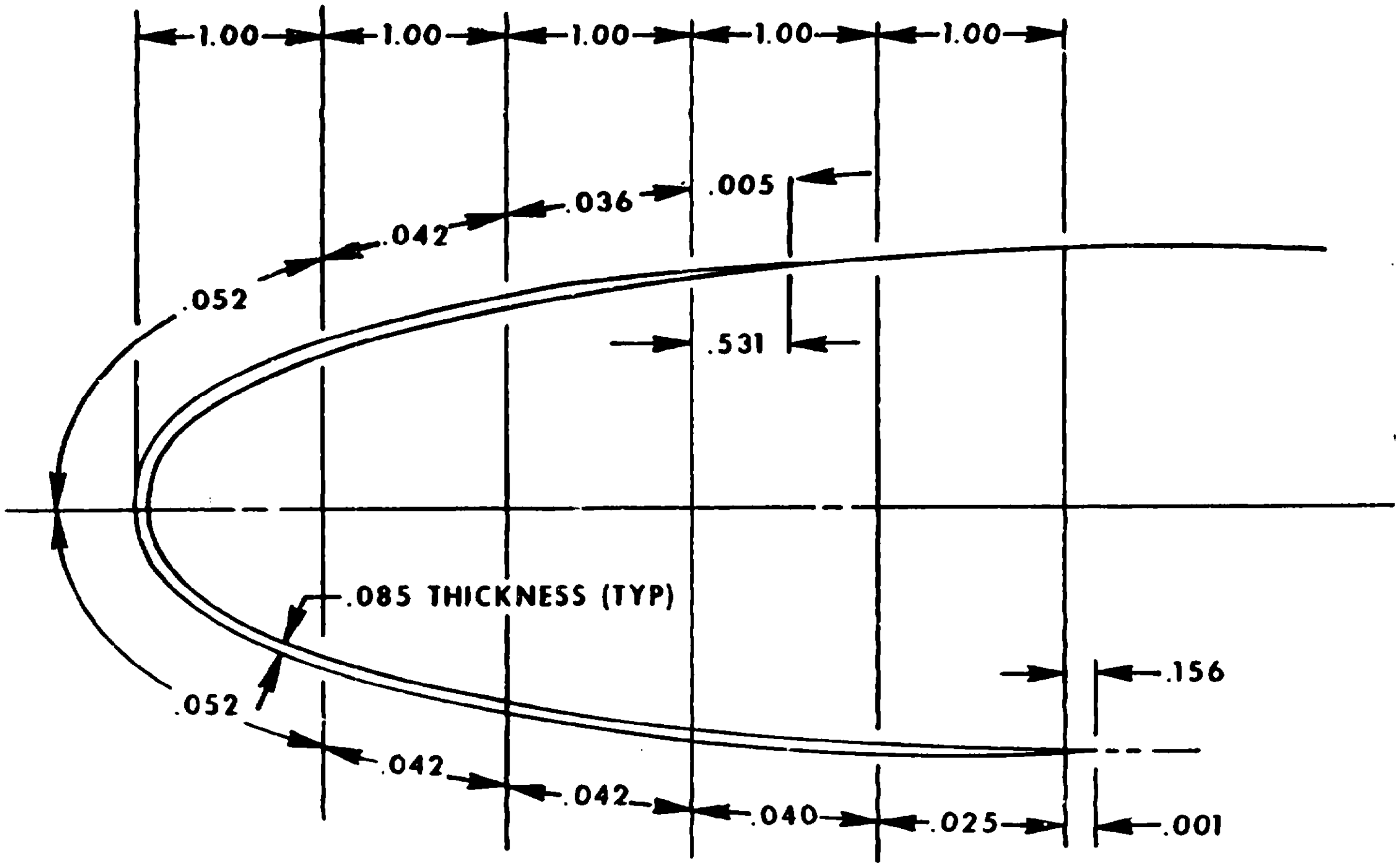


Figure 6. Pneumatic Deicer

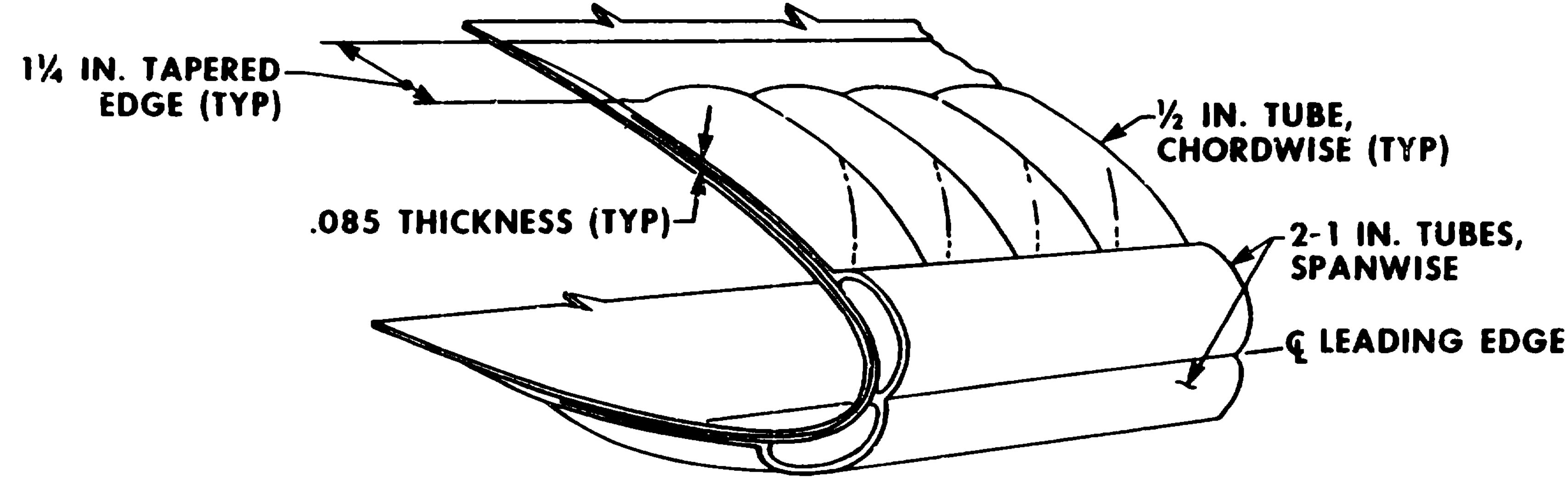


Figure 7. Pneumatic Deicer (Inflated Configuration)

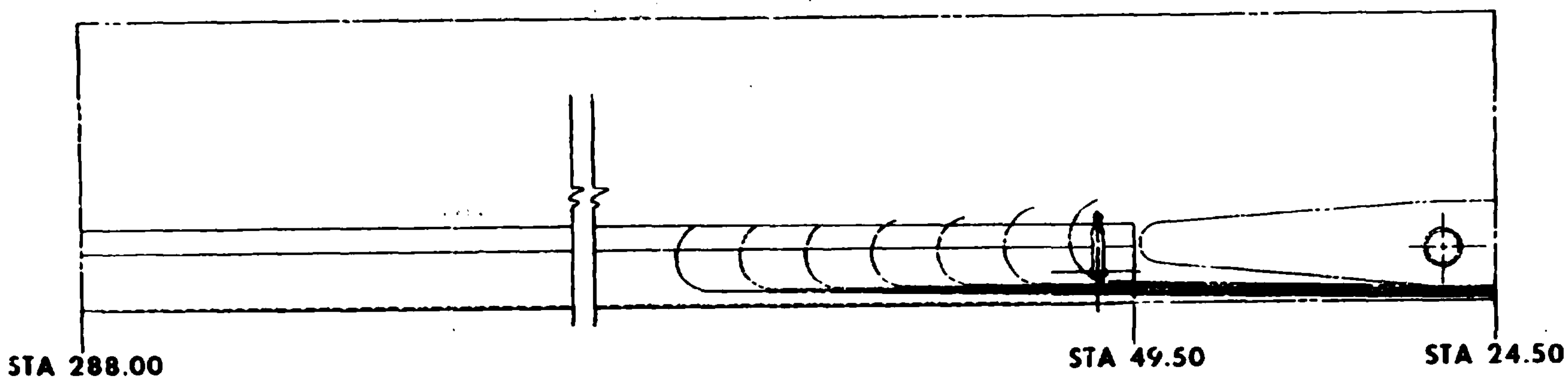


Figure 8. Deicer Installation

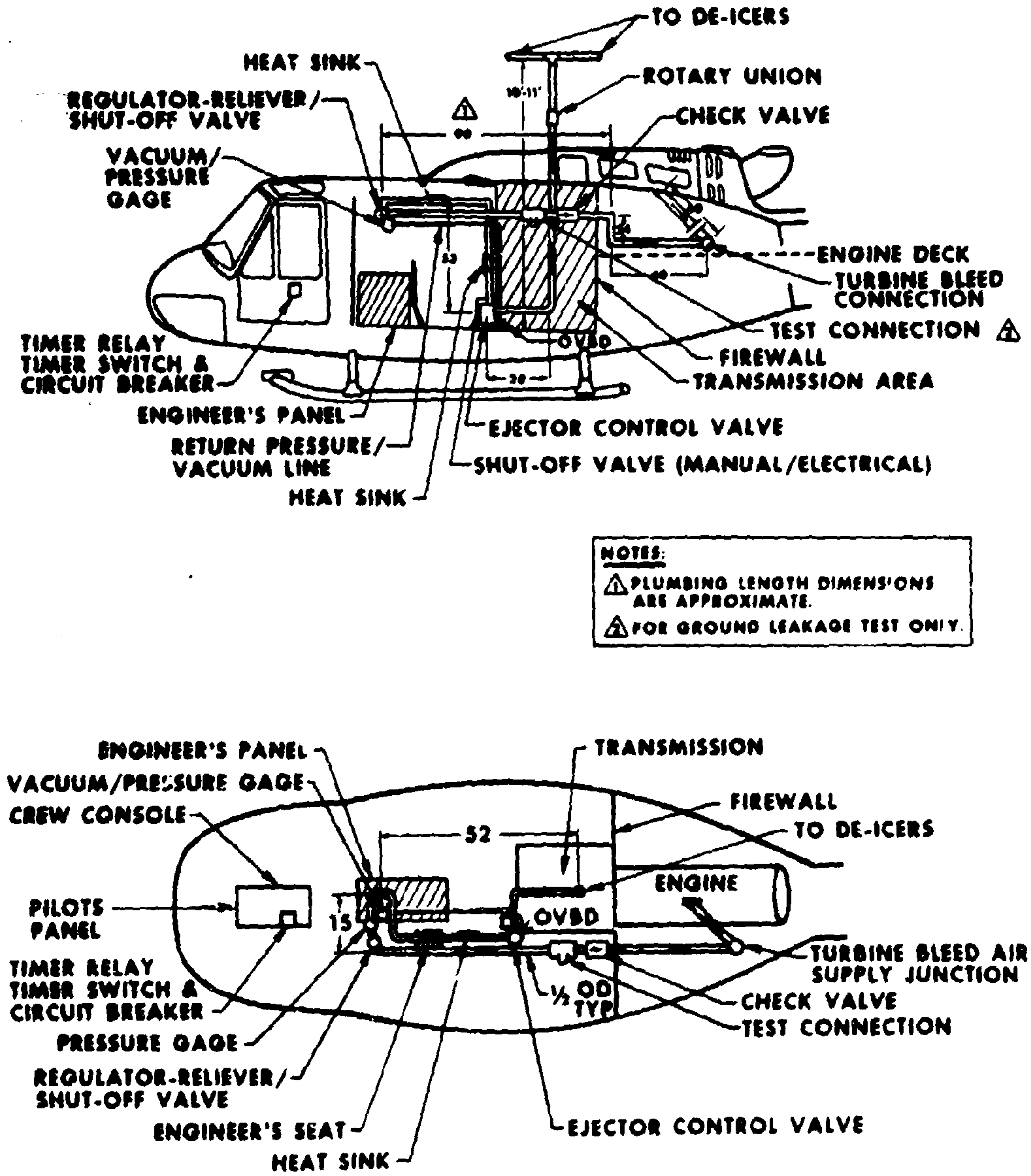


Figure 9. Schematic of Pneumatic Deicer System (Overview)

Check Valve

4. The check valve is a basic axial flow, spring loaded, poppet type unit. During normal operation, the valve poppet is open and has low resistance to flow of air from the engine compressor bleed air supply connection. The valve poppet closes to prevent leakage when the direction of air flow through the valve reverses during ground test only.

Regulator-Reliever/Shut-Off Valve

5. The regulator portion is a single-stage, diaphragm-type unit. When the regulated pressure reaches the regulator set point, the pressure on a diaphragm closes a port, shutting off inlet flow of air or preventing system pressure from rising above the system set point. The pressure-reliever is a separate spring-loaded poppet connected to a common system pressure port. The reliever is set to open and relieve pressures slightly above the system nominal pressure level. Modifications were made to set the regulated pressure to 25 PSI and the reliever to open at 27 PSI. The regulator-reliever/shut-off valve is shown in photos 3 and 4.

Ejector Flow Control Valve

6. The ejector flow control valve is a three-way solenoid valve (photos 3 and 5) with the deicer connected to the common port. In the de-energized condition, the deicer port is connected to the exhaust port through an internal ejector. System air pressure is connected to the inlet port which has an orifice that operates the ejector to supply vacuum to the deicer. When electrical power is applied to the valve's direct acting solenoid, the valve mechanism shifts to shut-off vacuum supply air and to direct inlet air through the deicer port to inflate the deicer. When electrical power is removed from the valve solenoid, the valve mechanism shifts to connect the deicer port to the exhaust port and vacuum is reapplied to the deicer. Modifications were made to a 3D2331 ejector flow control valve to increase the vacuum to 18 inches of mercury with a 25 PSI inlet pressure.

Control Switch

7. The contol switch (photo 6) is a single pole, momentary contact, toggle switch with screw terminals. The contacts are rated for 15 amps @ 125 VAC or 10 amps @ 250 VAC. This switch starts the timer for the single deicer inflation period.

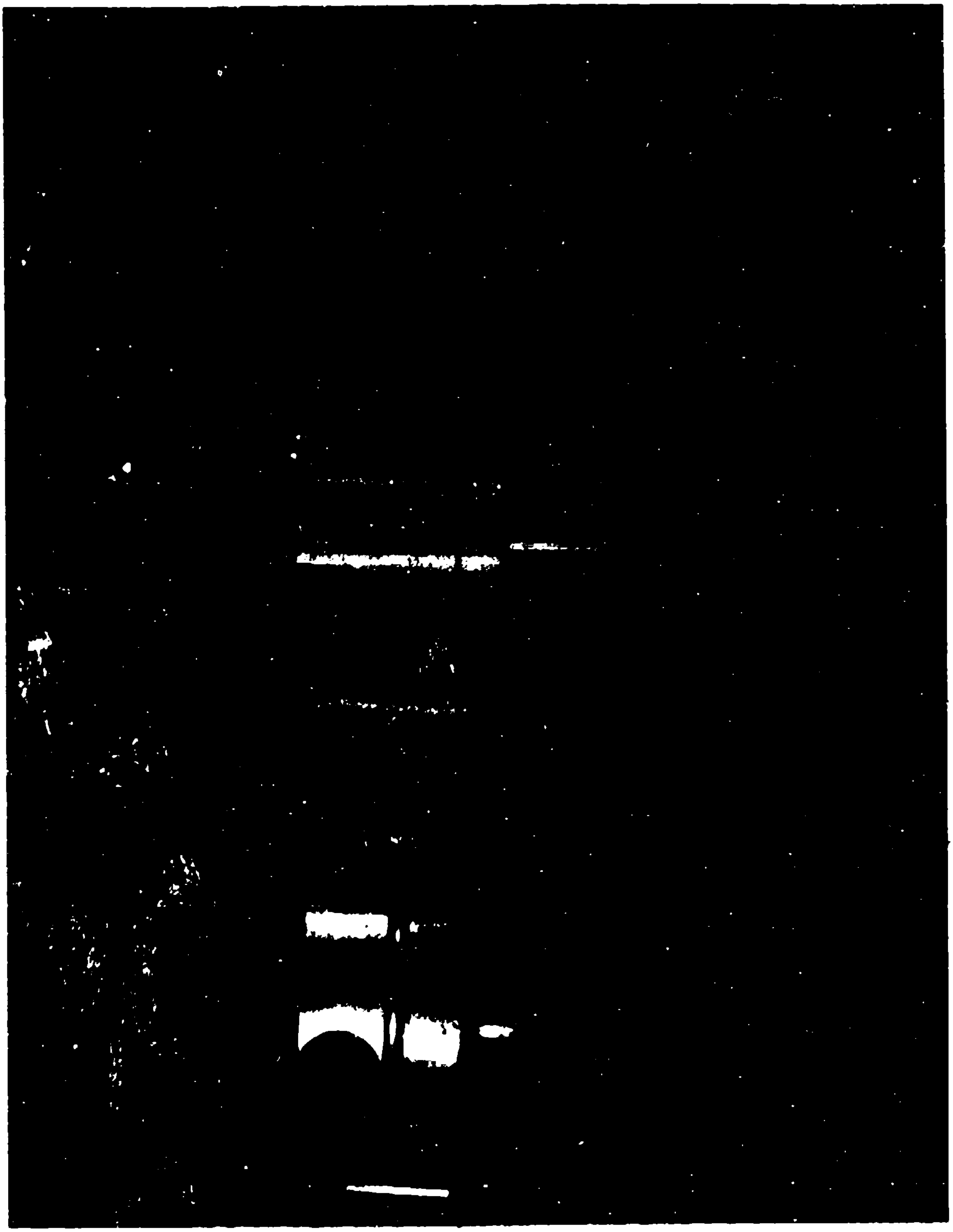
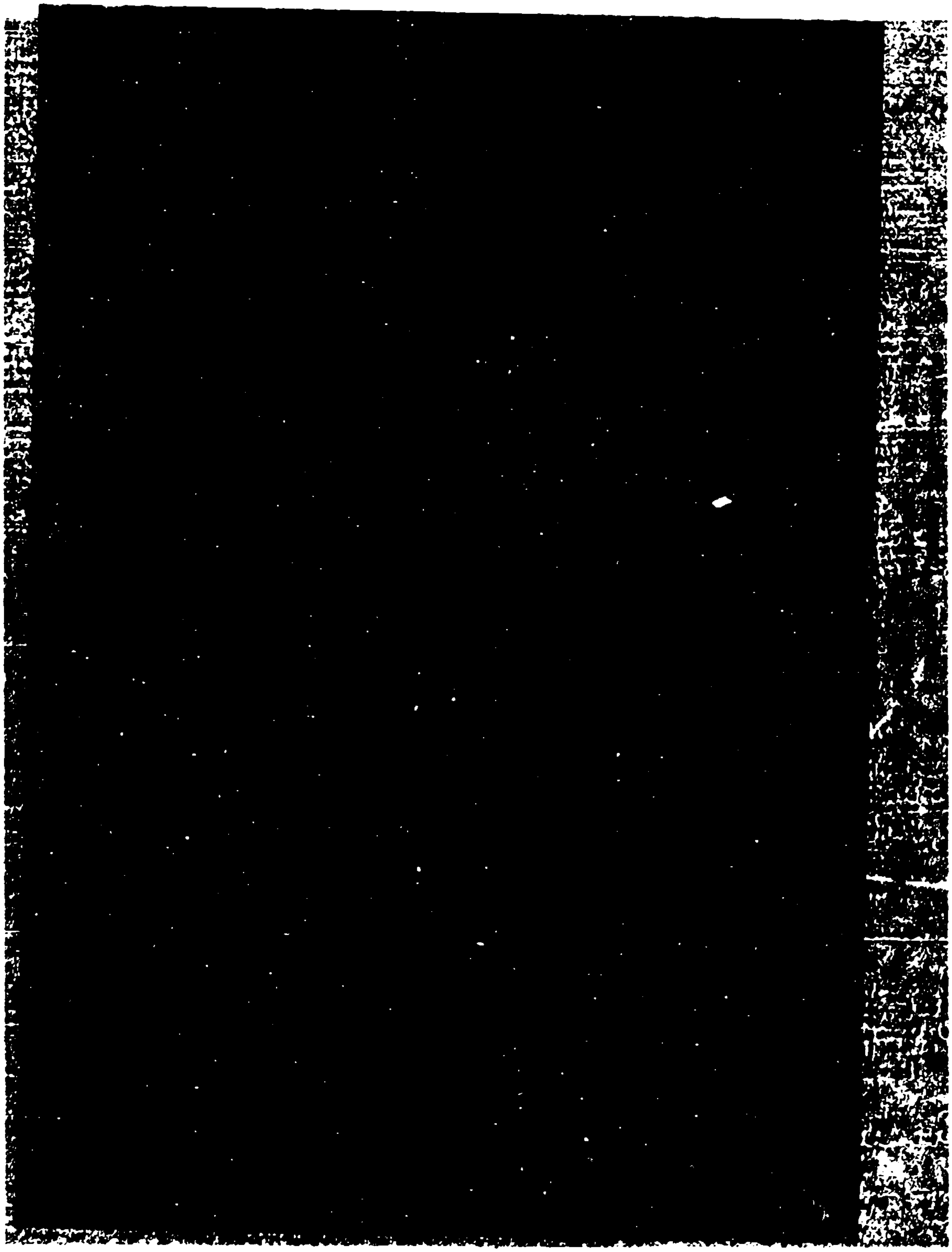


Photo 3. PBDS Components



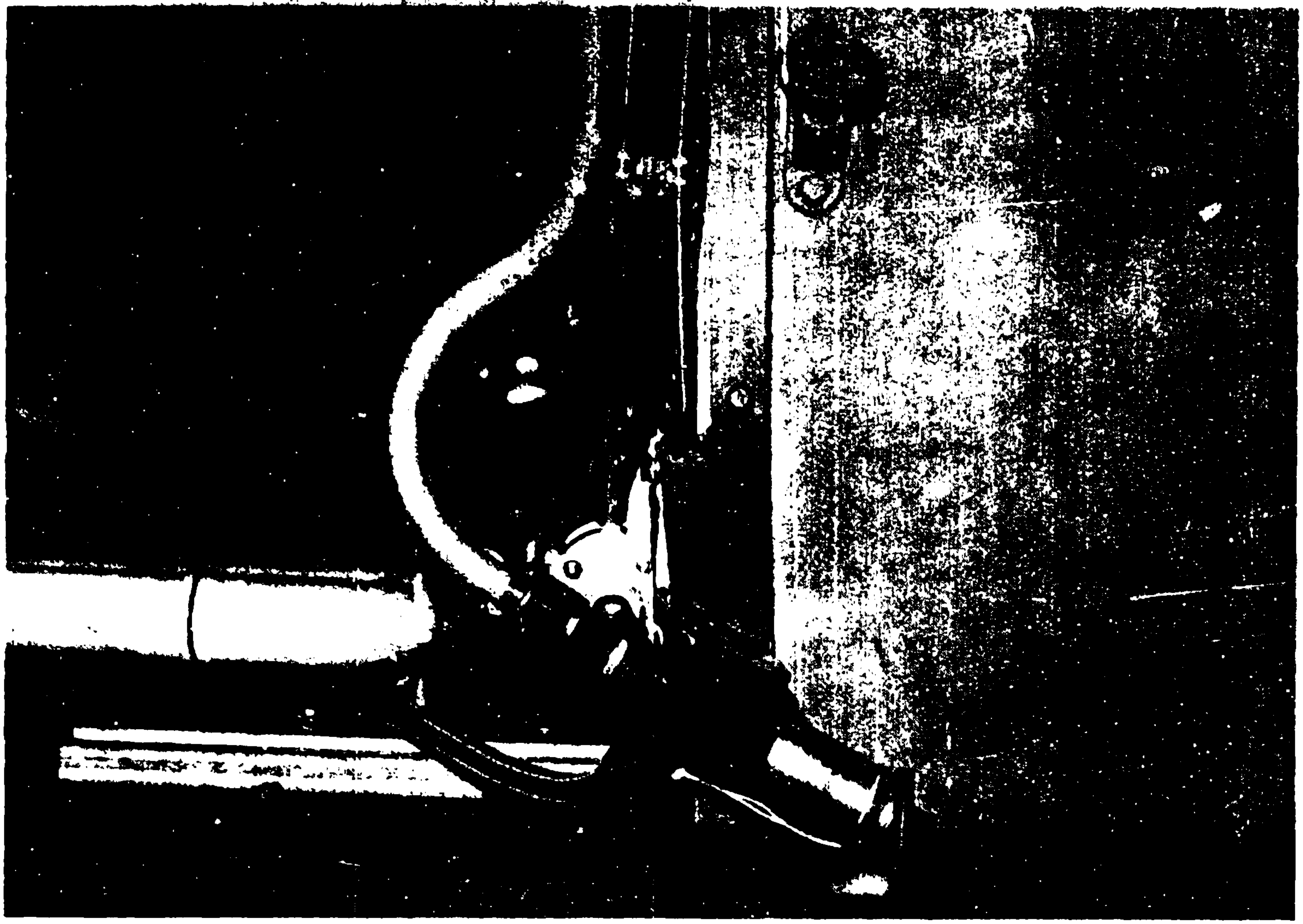


Photo 5. Ejector Flow Control Valve

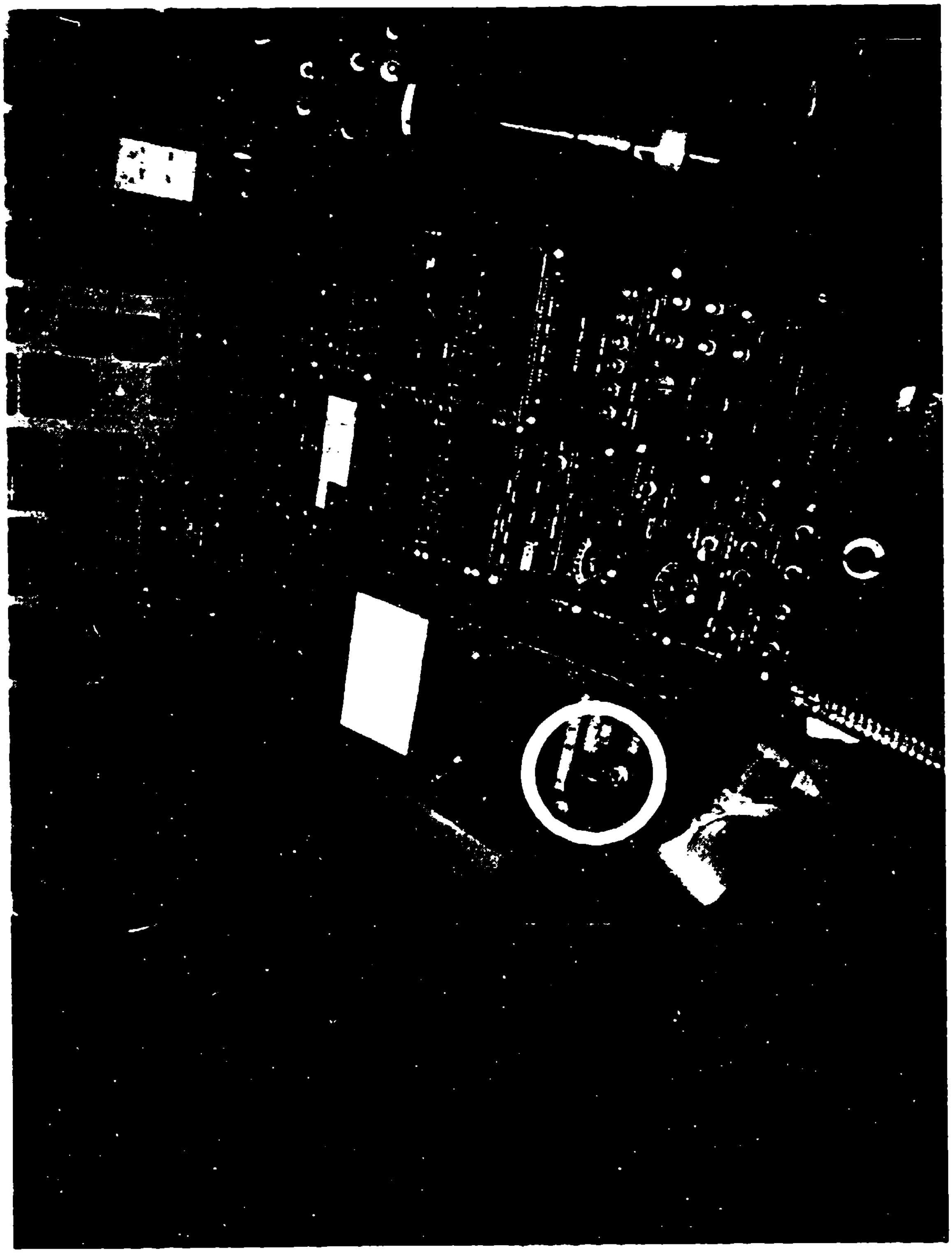


Photo 6. Control Switch

Rotating Union

8. The rotating union (photo 3) is a single pass, straight through air union, with two single row ball bearings. The union utilizes a balanced carbon steel to carbon graphite floating seal with "O" ring. The union is rated by the manufacturer at 1000 RPM and 150 PSI maximum.

Deicer Hose and Flap

9. The deicer hose (B-118) is a wire reinforced neoprene and fabric construction. The hose is modified with a flap constructed of rubber with fabric covering to top and bottom. The flap is used to attach the hose to the rotor blade surface. The deicer hose is attached to the drag link using nylon cable ties (TY-RAP, P/N TY-527M, Per MIL-S-23190 and MIS17332).

Pneumatic Deicer Timer

10. The pneumatic deicer time is an electrical-mechanical timing device (photo 3) utilizing a relay to provide a single timed output of electrical power to the ejector flow control valve solenoid. When the timer is actuated through the control switch the solenoid in the ejector flow control valve controlling air flow is immediately energized for the inflation period. At the end of the deicer inflation period, the solenoid is de-energized allowing the air to be evacuated from the deicers.

11. If the control switch is depressed for more than the preset deicer inflation period, the timer will de-energize at the end of the period and the control switch must be released and depressed again to start another inflation period.

12. Two modifications were made to the BFG time module: (1) time period was made adjustable from 0.5 to 3.5 seconds, and (2) the module was put in a case for environmental protection.

Pneumatic Deicer

13. The pneumatic deicer consists of a smooth rubber and fabric blanket containing two small spanwise deicing tubes along the leading edge, with the balance of the deicer consisting of smaller chordwise deicing tubes as shown in figure 6. All tubes in each deicer are simultaneously inflated through a single air connection located on the "breeze side" of the deicer. The deicer is cement-bonded to the airfoil leading edge.

14. Deicing action is provided by pressurizing the stretchable deicing tubes with compressed air. The inflation of the tubes produces bending and shearing stresses in the ice causing it to be broken into pieces and break its bond with the deicer surface. The scavenging effect of the air stream and centrifugal forces then removes the ice particles.

15. When the ejector flow control valve is de-energized, the vacuum is applied to the deicer tubes. This is necessary to resist negative aerodynamic pressures and to maintain the tubes in a flat or deflated condition. The "breeze side" of the deicer incorporates a 0.025-inch ply of "ESTANE®" (BFG polyurethane) to resist weathering and abrasion. To aid application to the blade, the deicer centerline leading edge reference line on the back of each deicer is to coincide with the centerline leading edge of each airfoil. The deicers are designed to operate at 25 PSI (nominal) and to be installed over the basic airfoil.

Pneumatic Deicer Modifications

16. All of the inflatable tube area was reportably designed with in internal self-venting construction that allows vacuum to be applied to all the tube area regardless of the point of applied vacuum (air connection). Analysis of deicer deflation time intervals pointed to a need for auxiliary venting to improve internal flow of air during deicer deflation. Four boot modifications were accomplished during this evaluation. The first BFG modification was to add auxiliary internal vents at the aft edges of the deicing tubes on the upper and lower surfaces as shown in figure 10. This modification added an increase in local profile thickness of about 0.04 inch; however, further testing did not show significant reductions in deflation time and the auxiliary vent appeared to become clogged with what BFG called soapstone deposits. (Soapstone is used during deicer production.) The second modification was applied to the inboard end of the deicers near blade station 51. The inboard ends of the deicer were temporarily debonded so that an additional air path could be made from the area of the air connection to the auxiliary air vents at the aft edges of the tube area as shown in figure 10. The added air path appeared as a full width fabric patch and added about 0.06-inch thickness which improved deflation time slightly. The third modification required replacement of the deicers and consisted of the addition of auxiliary venting to the full length of both one-inch spanwise deicing tubes located at the blade leading edge as shown in figure 10. This reduced deflation time to 40 to 50 seconds. The deicer thickness at the span tubes increased approximately 0.04 inch and tapered to the nominal thickness of 0.85 inch over a 1.25-inch distance aft of

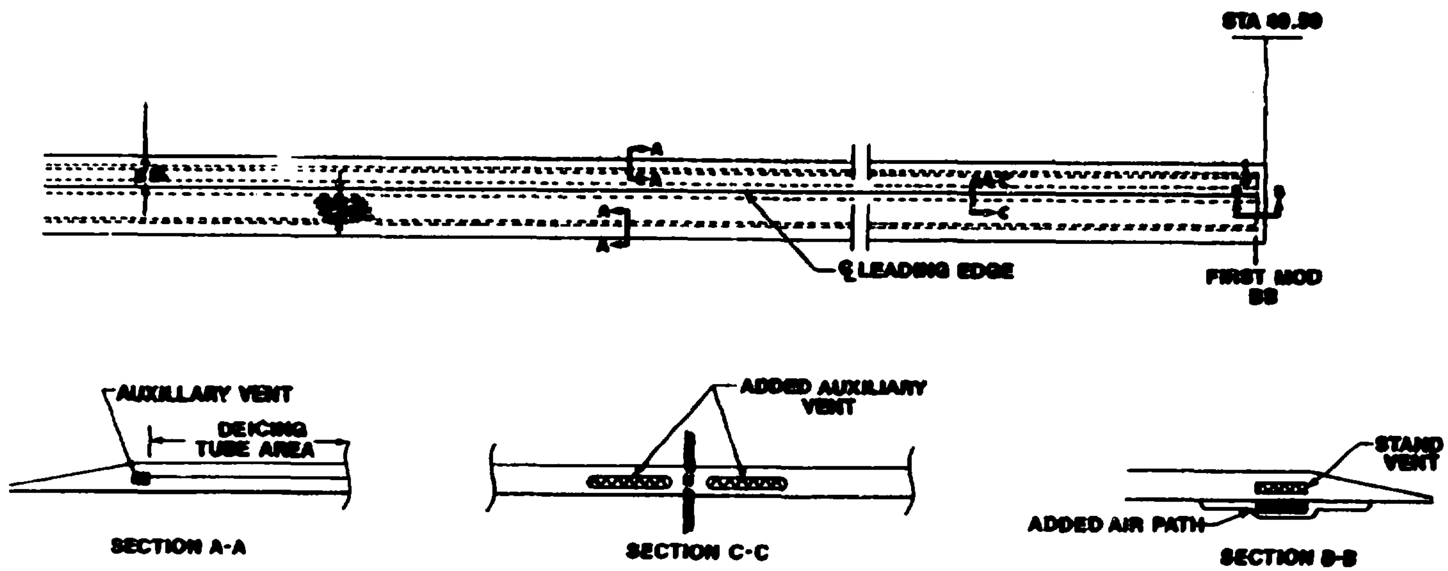


Figure 10. Pneumatic Deicer Modifications

the tubes on the upper and lower surfaces. The fourth modification included the addition of auxiliary venting in the spanwise tubes but removed the auxiliary air path aft of the deicer tubes. This modification did not effect the deflation rate.

APPENDIX C. INSTRUMENTATION AND SPECIAL EQUIPMENT

INSTRUMENTATION

1. The test instrumentation was installed, calibrated, and maintained by the US Army Aviation Engineering Flight Activity (USAAEFA), except the load strain gages which were installed by Bell Helicopter Textron (BHT). Digital and analog data were obtained from calibrated instrumentation and were recorded on magnetic tape and/or displayed in the cockpit. The digital instrumentation system consisted of various transducers, signal condition units, an eight-bit pulse code modulation (PCM) encoder, and an Ampex AR 700 tape recorder. Time correlation was accomplished with a pilot/engineer event switch and on-board recorded and displayed Inter-Range Instrumentation Group (IRIG) B time. Analog data were recorded on one track of the AR 700 recorder through the use of a voltage control oscillator. Various specialized test indicators displayed data to the crew continuously during the flight. The instrumentation rack is shown in photo 1.

2. In addition to standard ship's instruments, the following parameters were displayed on calibrated test instruments and recorded manually in the cockpit:

Pilot's Panel

- Airspeed (ship's)
- Altitude (ship's)
- Fuel flow
- Fuel used
- Engine torque
- Engine inlet screen differential pressure
- Rosemount outside air temperature (OAT)
- Rosemount liquid water content (LWC)
- Tether cable tension

Engineer's Panel

- Main rotor speed
- Ejector control valve regulated pressure.
- Vacuum/pressure of the deicer boot
- Engine bleed air temperature at engine deck
- Engine bleed air temperature at the ejector control valve
- Engine bleed air temperature at the regulator-reliever/shut-off valve

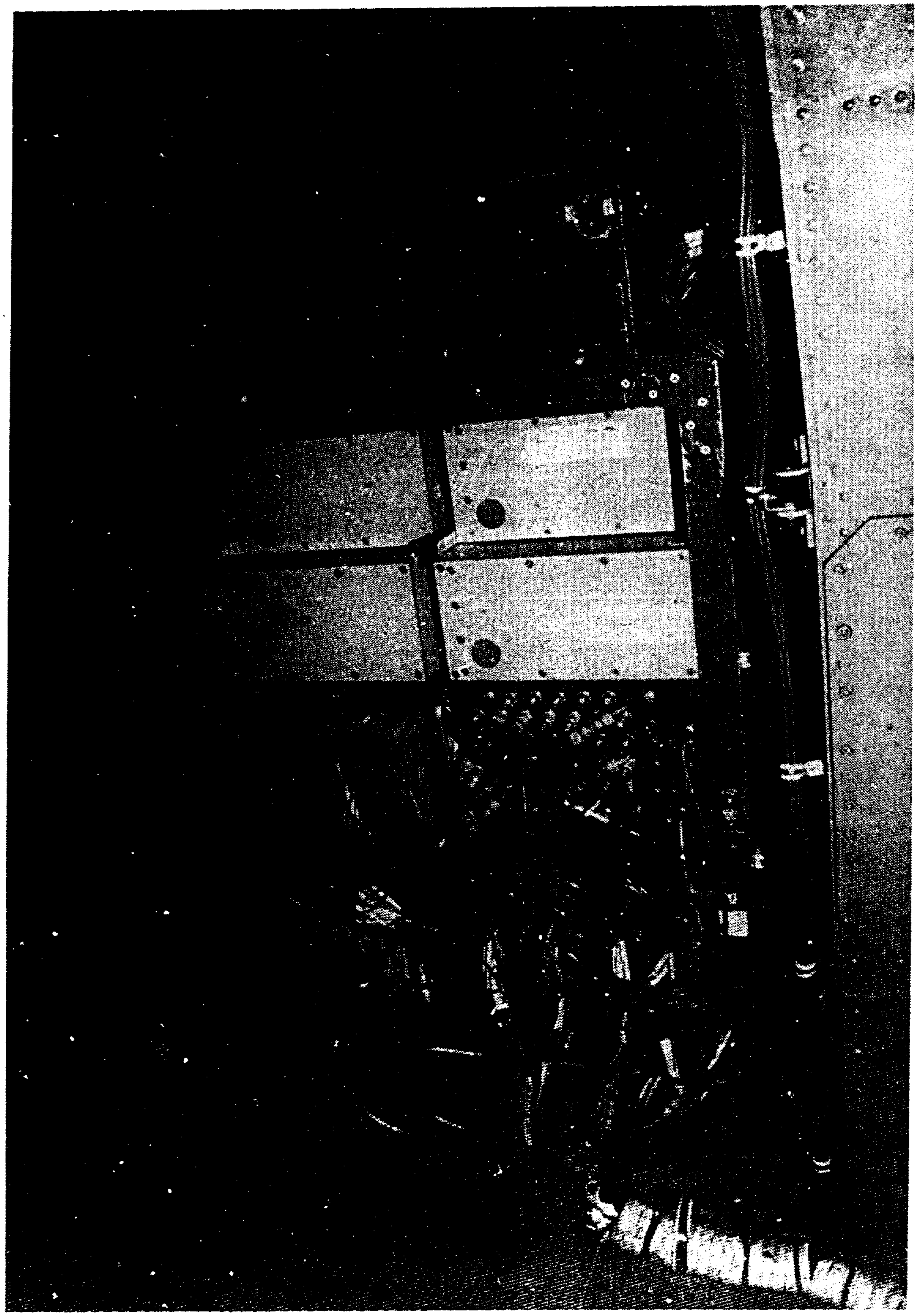


Photo 1. Instrumentation Rack

3. The following parameters were recorded on magnetic tape.

PCM Parameters

Control position
- Longitudinal
- Lateral
- Directional
- Collective

Engine torque
Fuel flow
Fuel used
Airspeed (ship's)
Altitude (ship's)
Main rotor speed
Bleed Air Press (pressure/vacuum)
Bleed Air Press for PBDS
Aircraft attitude
- Pitch
- Roll
- Yaw

Aircraft rates
- Pitch
- Roll
- Yaw

Outside air temperature
Center of gravity normal acceleration
Tether cable tension
Pilot seat vertical vibration
Main rotor blade pitch angle (red blade)
Main rotor blade flapping angle (red blade)
Main rotor azimuth
Main rotor mast torque
Main rotor mast bending perpendicular
Main rotor blade beam and chord bending at the following stations
- Station 35
- Station 84
- Station 150
- Station 192
- Station 234

4. The following frequency modulated parameters were recorded and telemetered to a ground station for real-time safety of flight monitoring.

Main rotor pitch link axial force
Main rotor mast bending parallel
Main rotor blade beam bending at station 192

Main rotor blade chord bending at station 192
Main rotor hub beam bending at station 6.3
Main rotor hub chord bending at station 6.3

SPECIAL EQUIPMENT

Icing Spray Rig

5. The icing spray rig, operated by the National Research Council of Canada, is located adjacent to Canadian Forces Base Uplands in Ottawa, Ontario, Canada. It consists of a 75 foot by 15 foot nozzle array on a 59 foot mast. The spray assembly consists of 156 nozzles through which water and steam are pumped. Water atomization is achieved through the use of the steam. The spray rig has two operating heights, full up or full down (approximately 54 feet and 23 feet from the ground to the top of the spray array, respectively). Wind anemometer cups are attached to the top of the nozzle array to provide windspeed to a meter located in the control room.

High Speed Video System

6. To document ice accretions and deicing action of the PBDS during icing spray rig tests, an NAC, Incorporated High Speed Video System HSV-200 was used. The system consisted of a color camera, portable record/playback video tape recorder unit, and a monitor. The color camera contains a built-in mechanical shutter for use in ambient light which limits exposure to 500 microseconds per field in order to eliminate motion blur. The system utilizes a VHS standard video format with a scanning rate of 200 fields per second and a maximum recording time of 36 minutes with VHS-T120 cassette. Playback modes include: normal play at the recorded speed; slow motion forward continuously variable from 1 to 15 fields/second; slow motion reverse at 10 fields/second; single frame advance; and still, stop motion (45 second duration).

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

PHASE I FLIGHT LOADS SURVEY

1. Reference 11, appendix A was used as a guide for evaluation purposes. Critical loads parameters indicated in appendix C were telemetered to a ground station for real-time safety of flight monitoring by a Bell Helicopter Textron structures engineer.

2. Initial testing was conducted with the aircraft ballasted to 9500 pounds gross weight and tethered to a "deadman" anchor. Rotor system loads associated with the three Pneumatic Boot Deicing System (PBDS) configurations (deflated, inflated, and vented) were monitored via telemetry and analyzed during starting and acceleration to engine idle, normal operating rpm, and maximum governing rpm. Since no problems were encountered, power was increased in increments of 5 PSI of engine torque to allowable limits at rotor speeds of 294 and 324 revolutions per minute (rpm). Dynamic system-engine compatability was evaluated at each of the two rotor speeds at three power settings corresponding to minimum, mid-range, and maximum. This evaluation was accomplished by manually cycling the collective and directional controls at the critical frequency of the dynamic system (approximately 2.9 Hertz) and at frequencies 0.1 Hertz slightly above and below critical.

3. After analysis of the ground test loads survey data by the structures engineer, inflight testing was accomplished. The critical loads parameters were monitored and analyzed during hover, low-speed, level flight, climbs and descents, and autorotational entries. Inflight dynamic system-engine compatibility was also evaluated.

PHASE II PERFORMANCE AND HANDLING QUALITIES

Performance

4. The helicopter performance test data were generalized by use of nondimensional coefficients and were such that the effects of compressibility and blade stall were not separated and defined. The following nondimensional coefficients were used to generalize the hover and level flight test results obtained during this flight test program.

a. Coefficient of power (C_P):

$$C_P = \frac{SHP(550)}{\rho A(\Omega R)^3} \qquad (1)$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{\text{Thrust}}{\rho A(\Omega R)^2} \qquad (2)$$

c. Advance ratio (μ):

$$\mu = \frac{1.6878 V_T}{\Omega R} \qquad (3)$$

Where:

SHP = Engine output shaft horsepower
550 = Conversion factor (ft-lb/sec/shp)
ρ = Air density ($slug/ft^3$)
A = Main rotor disc area (ft^2) = 1809.56
Ω = Main rotor angular velocity (radian/sec) = 33.93 (at 324 rpm)
R = Main rotor radius (ft) = 24
Thrust = Gross weight (lb) during free flight in which there is no acceleration or velocity component in the vertical direction. Tether load must be added in the case of tethered hover.
1.6878 = Conversion factor (ft/sec/knot)
V_T = True airspeed (knot)

For a rotor speed of 324 rpm, the following constants were used:

$A = 1809.56\ ft^2$
$\Omega R = 814.30\ ft/sec$
$A(\Omega R)^2 = 1199893575\ ft^4/sec^2$
$A(\Omega R)^3 = 9.770743174 \times 10^{11}\ ft^5/sec^3$

<u>Shaft Horsepower Required</u>

5. The engine output shaft torque was determined from the engine manufacturer's torque system. The output shp was determined from the engine output shaft torque and rotational speed by the following equation:

$$SHP = \frac{2\pi \times N_p \times Q}{33{,}000} \qquad (4)$$

Where:

N_p = Engine output shaft rotational speed (rpm)
Q = Engine output shaft torque (ft-lb)
33,000 = Conversion factor (ft-lb/min/shp)

Hover Performance

6. Hover performance data were gathered during 5-foot tethered hovering flight. Power was varied between data points from the minimum required to maintain tension in the tether cable, to the maximum power available. Cable tension was measured and added to the aircraft gross weight to determine thrust (required in equation 2). To further increase the range of C_T and C_P, main rotor speed was varied from approximately 91 to 100 percent.

Level Flight Performance

7. Level flight performance data were reduced using equations 1, 2, and 3. The speed power was flown at a predetermined constant C_T by maintaining a constant gross weight to density ratio (W/σ). The aircraft was flown in coordinated (ball-centered) flight with altitude increased between data points to maintain the constant W/σ.

8. Test-day (measured) level flight power was corrected to standard-day conditions by assuming that the test-day dimension less parameters C_{P_t}, C_{T_t}, and μ_t, are identical to C_{P_s}, C_{T_s}, and μ_s, respectively.

From equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \frac{\rho_s}{\rho_t} \tag{5}$$

Where:

Subscript t = test day
Subscript s = standard day

HANDLING QUALITIES

9. Stability and control data were collected and evaluated using standard test methods as described in reference 13, appendix A.

10. The Handling Qualities Rating Scale (HQRS) presented in figure 1 was used to augment pilot comments relative to handling qualities and work load.

Vibration Rating Scale

11. The Vibration Rating Scale (VRS) presented in figure 2 was used to augment crew comments on aircraft vibration levels during PBDS activation cycles and main rotor asymmetric ice sheds encountered during Phase III testing.

PHASE III ICING SPRAY RIG TESTING

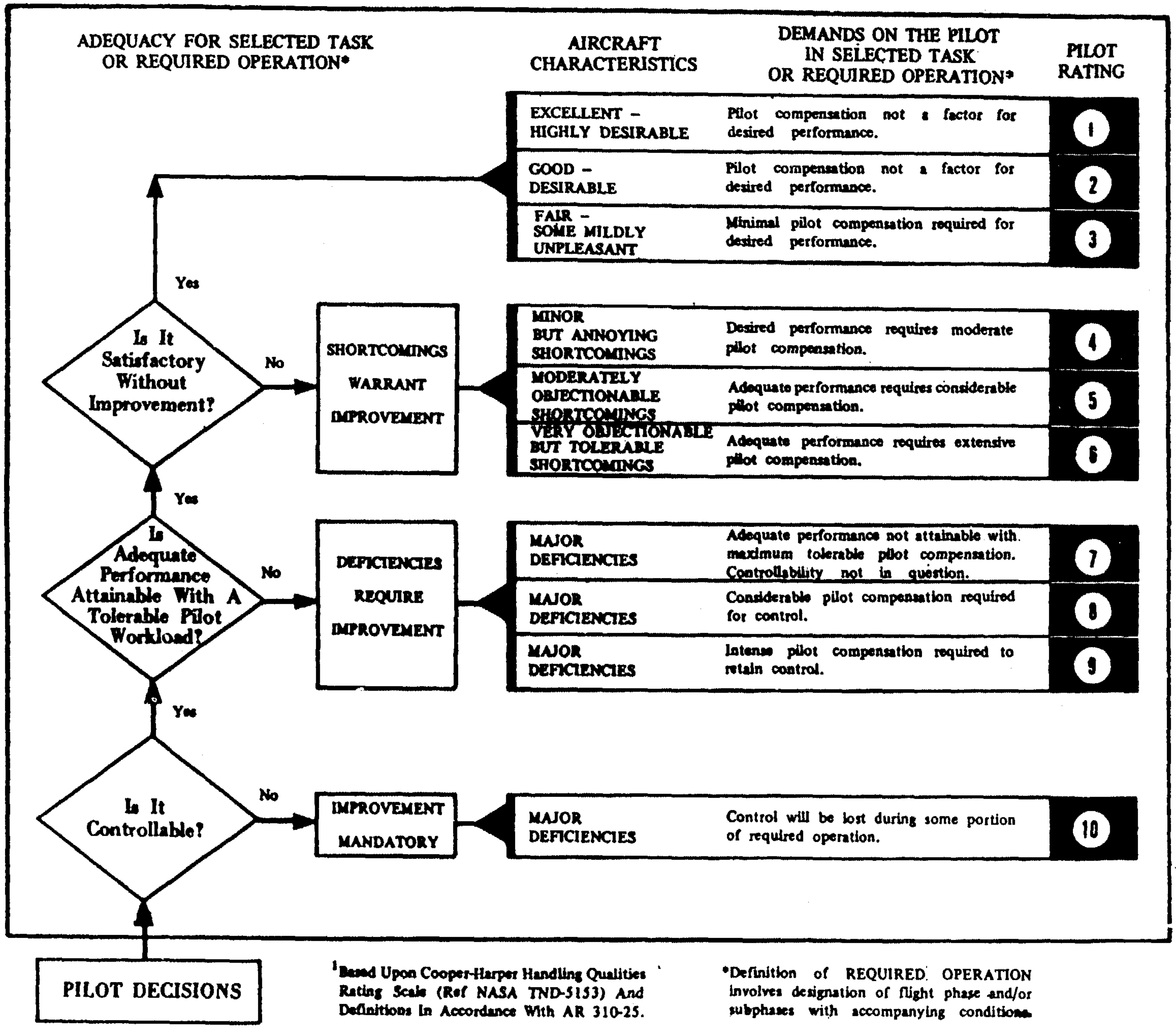
12. Phase III testing was conducted using the Icing Spray Rig described in appendix C. Standoff distance for the test aircraft was 100 feet measured from the rig mast to the aircraft main rotor mast (approximately 76 feet from blade tip to nozzle array for the UH-1H). Red marker panels were placed on the ground and personnel in the control room and in a ground vehicle equipped with an FM radio were used to advise the aircraft on proper position in the spray cloud. Proper cloud entry technique was to establish a 20 to 30 foot hover outside the cloud and transition into the cloud from the side.

13. The liquid water content (LWC) at the test aircraft is affected by the time of flight of the droplet and the gustiness of the wind. The time of flight of the droplet is determined by the wind speed and the distance from the spray array. Figure 3 shows the correction factor, based on gustiness, to be applied to the required test LWC in order to determine the LWC needed at the spray array. The gustiness is defined as the average value of variation from the mean wind speed, the three ranges being:

Low gustiness: 0 to $\pm$1 1/2 mph
Medium gustiness: $\pm$1 1/2 to $\pm$3 mph
High gustiness: $\pm$3 mph or greater

The correction factor was determined empirically by the National Research Council of Canada during flights of a Bell 47J helicopter in the cloud. The thickness of accreted ice on the rotor blades was measured and the LWC required to produce such an accretion at the conditions of flight was calculated.

14. Once the LWC necessary at the spray array is determined by multiplying the required test LWC by the correction factor obtained from figure 3, figure 4 is used to determine the operating water flow rate and steam pressure. After locating the mean



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure 1. Handling Qualities Rating Scale

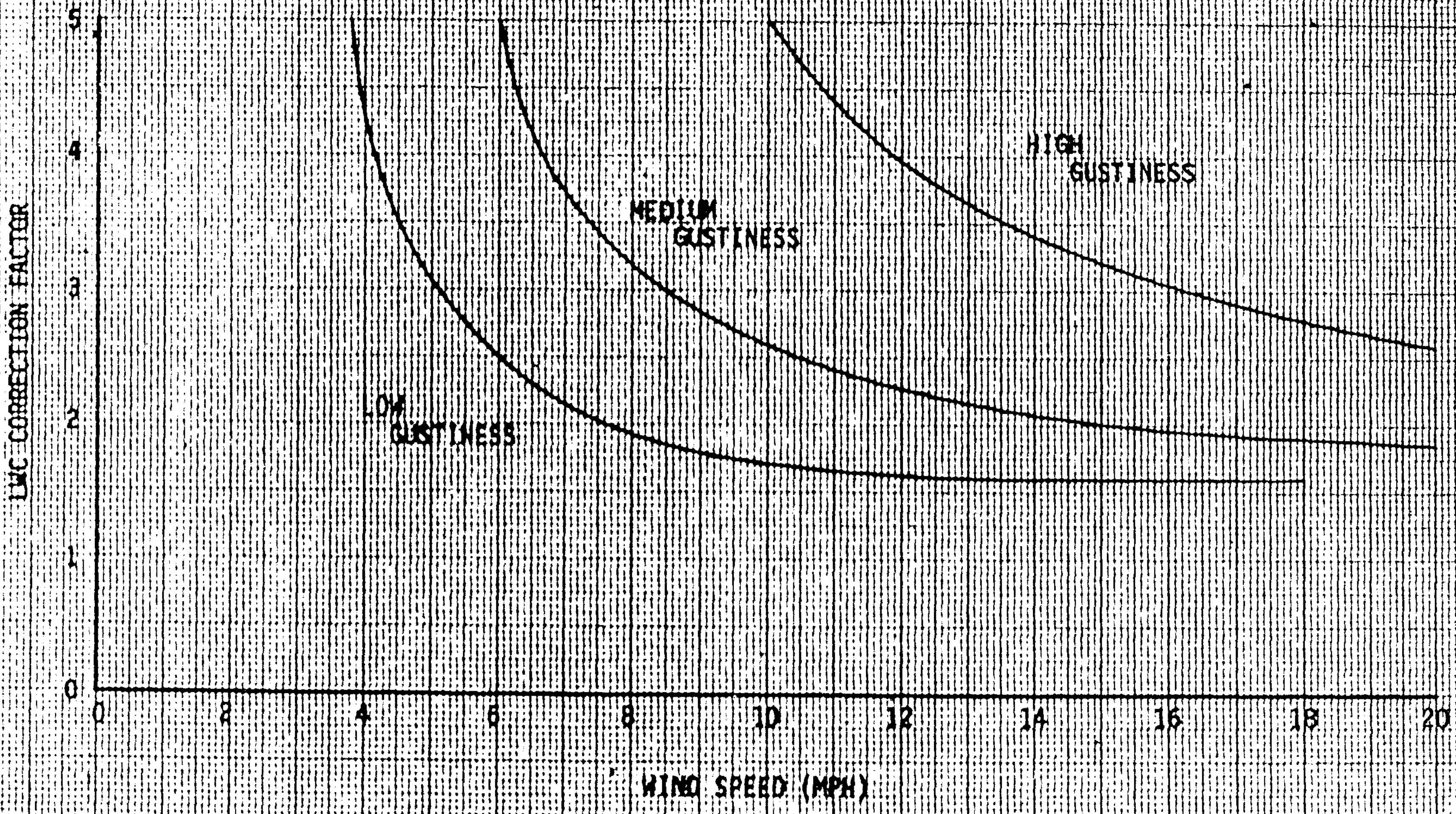
DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1, 2, 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4, 5, 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7, 8, 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 2. Vibration Rating Scale

FIGURE 3 CORRECTION FACTOR TO BE APPLIED TO CALCULATED LWC

NOTES:
1. HELICOPTER 100 FT FROM SPRAY ARRAY
2. LWC AT SPRAY ARRAY = (REQUIRED TEST LWC) X (LWC CORRECTION FACTOR)
3. GUSTINESS: LOW = 0 TO ± 1½ MPH
   MEDIUM = ± 1½ TO ± 3 MPH
   HIGH = ± 3 MPH OR HIGHER



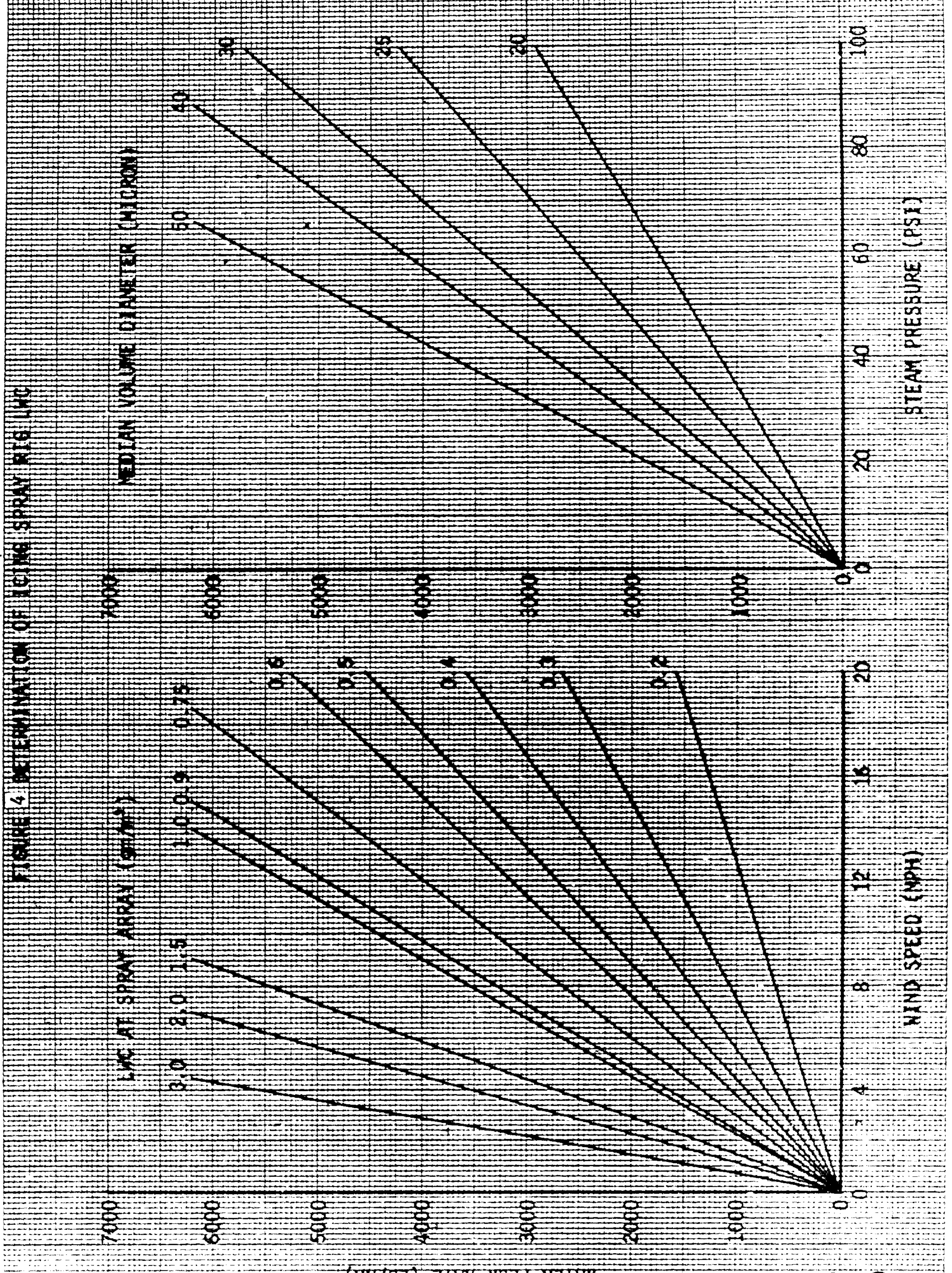


FIGURE 4 DETERMINATION OF ICING SPRAY RIG LWC

wind speed on the wind speed ordinate, move vertically to the curve of the necessary spray array LWC. From this point move horizonally to the right to intersect the desired test droplet diameter. The coordinates of this point give the required steam pressure and water flow rate.

15. The test procedure involved first immersing the aircraft in the spray cloud for a time interval estimated necessary to accrete a 0.25 inch ice thickness at the main rotor blade mid-span point. The aircraft was shutdown and these accretions were documented by hand measurement and still photography. The aircraft was restarted and a PBDS activation cycle was performed to shed the ice. The aircraft was again shutdown and an inspection of the ice remaining was made to document the effectiveness of ice removal. After photographic documentation, residual ice was manually removed, the aircraft was restarted, and the spray cloud was reentered for an immersion time necessary to reach either a predetermined aircraft operating limit or 15 minutes, whichever occurred first. The applicable aircraft limits for this test were either an increase of 5 PSI engine torque, an increase in inlet differential pressure of 10 inches of water, or onset of a moderate vibration level resulting from an asymmetric shed. When the aircraft exited the cloud, a PBDS activation cycle was performed. PBDS activation cycles were documented with the high-speed video system described in appendix C.

APPENDIX E. TEST DATA

FIGURE 1
NONDIMENSIONAL HOVERING PERFORMANCE
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM
SKID HEIGHT = 5 FEET

SYM	AVG OAT (°C)	AVG DENSITY ALTITUDE (FT)	CONFIGURATION
○	8.0	1800	SYSTEM DEFLATED
□	9.0	1900	SYSTEM VENTED
△	9.5	1980	SYSTEM INFLATED

NOTES: 1. TETHERED HOVER TECHNIQUE
2. WINDS LESS THAN 5 KNOTS

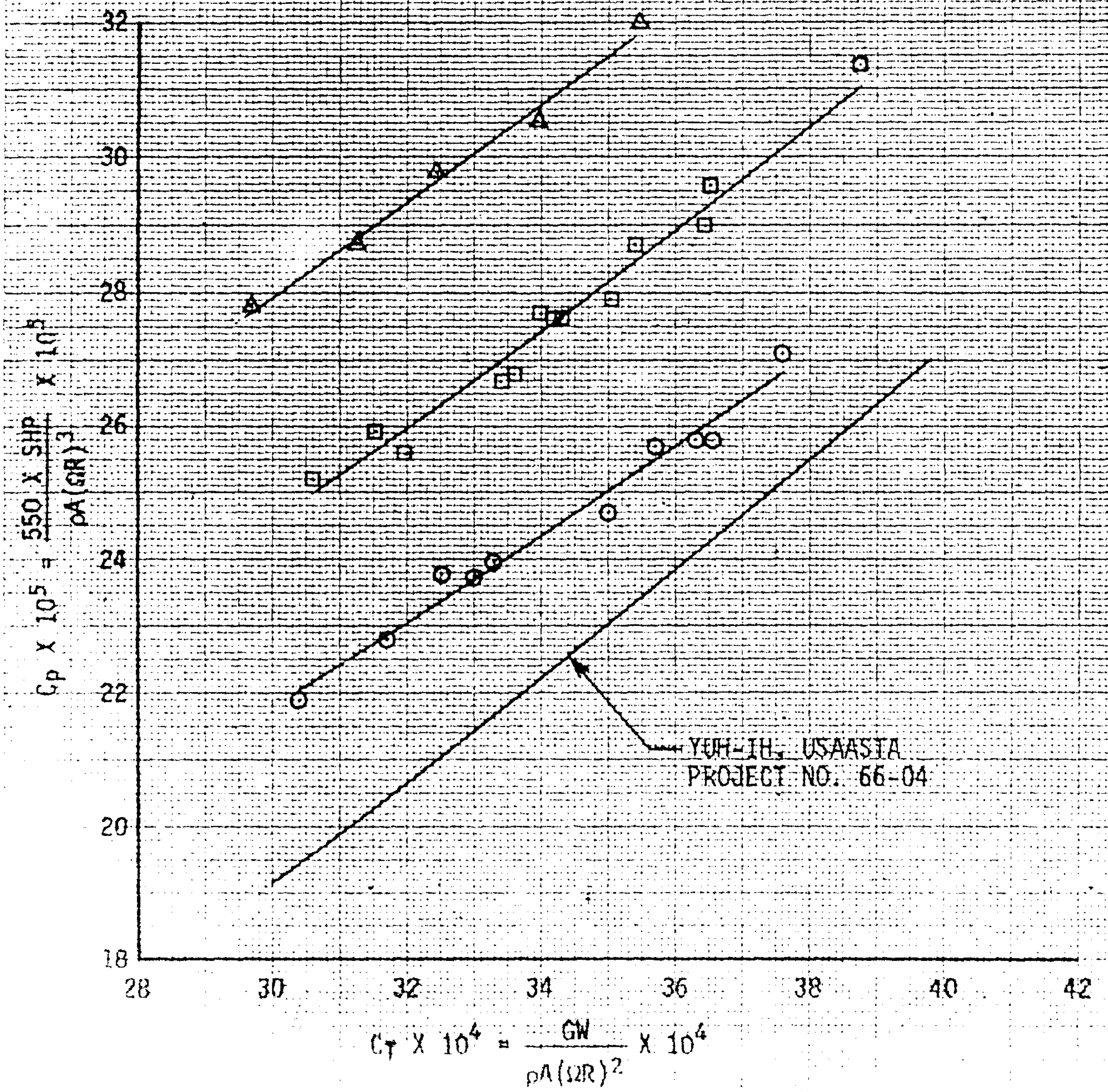


FIGURE 2
LEVEL FLIGHT PERFORMANCE
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DE-ICE SYSTEM

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG C_T	FBDS CONFIG
○	[illegible]	135.8	8820	2.5	324	0.003633	DEFLATED
□	[illegible]	134.8	9100	0.5	324	0.003644	VENTED

NOTE: BALL CENTERED

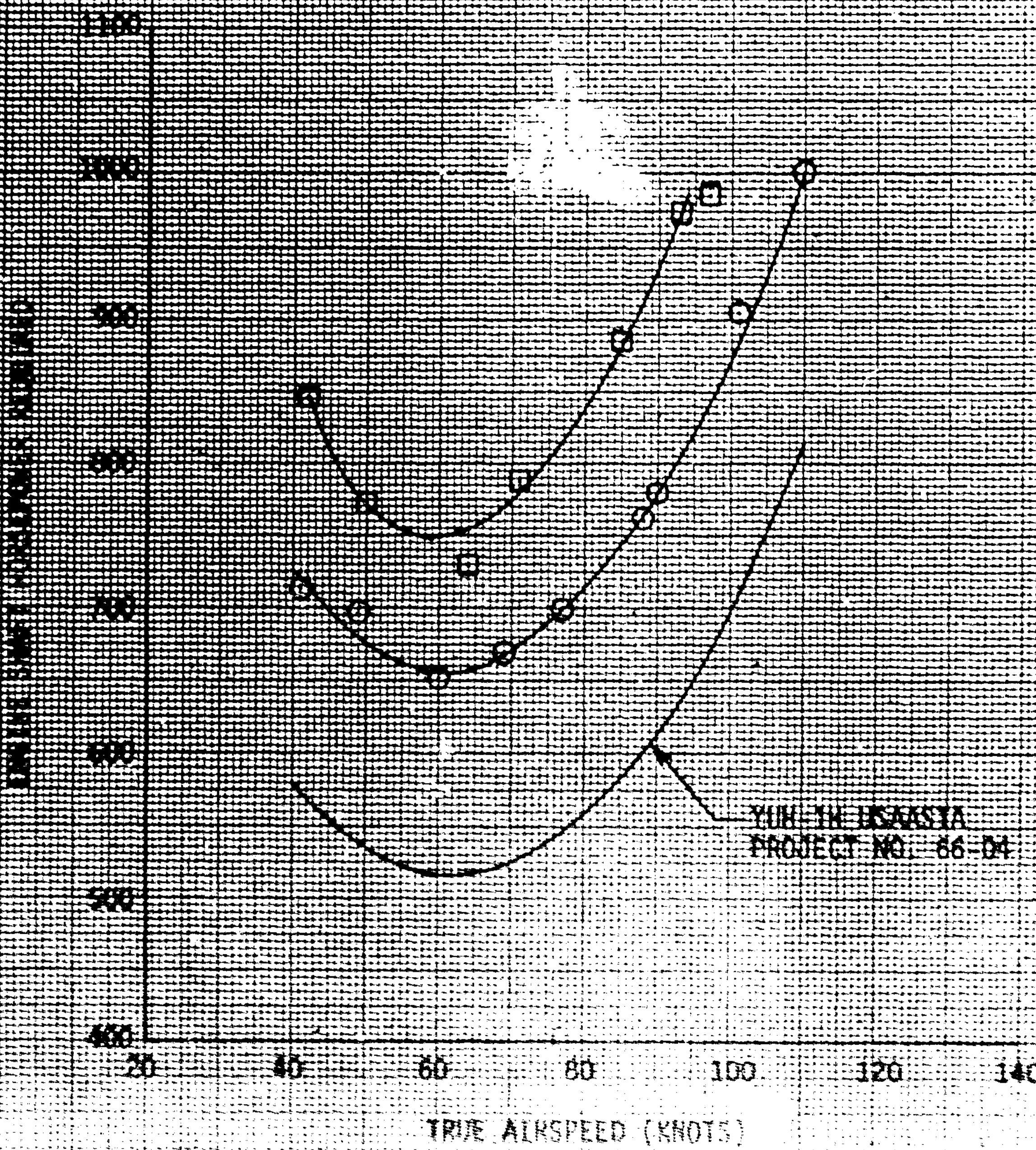


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FIGURE 3

PBDS ACTIVATION CYCLE

JUH-1H USA S/N 70-16318

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	TRIM DENSITY ALTITUDE (FEET)	AVG OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	FLIGHT CONDITION	CONFIGURATION
7400	136.1 (MID)	0.5	3900	6.0	322	81	LEVEL	PBDS

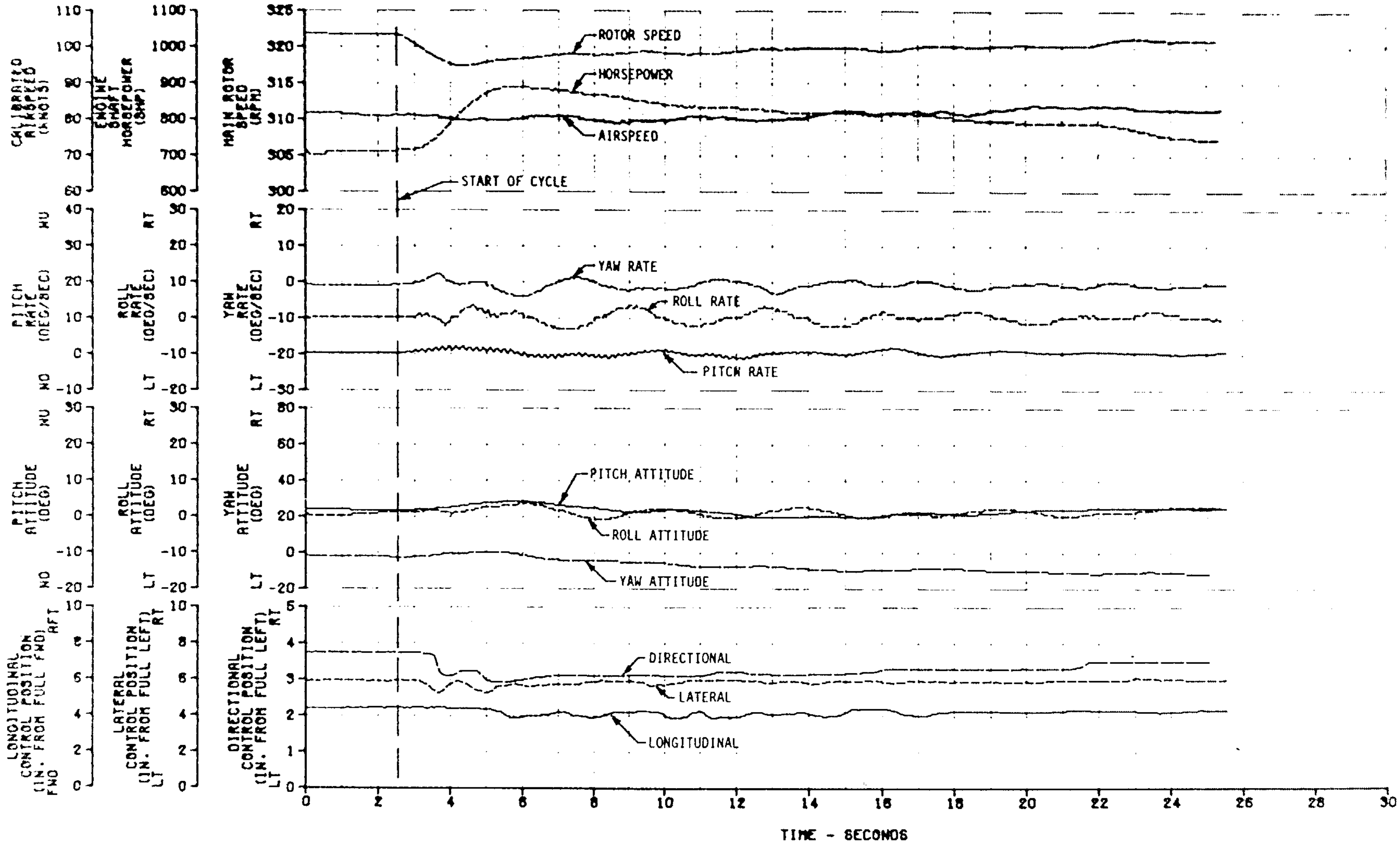

FIGURE 4
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
JUH-1H USA S/N 70-16318

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	PBDS CONFIGURATION
7800	137.8 MID	0.5	5120	7.5	324	LEVEL	DEFLATED

NOTE: BALL-CENTERED

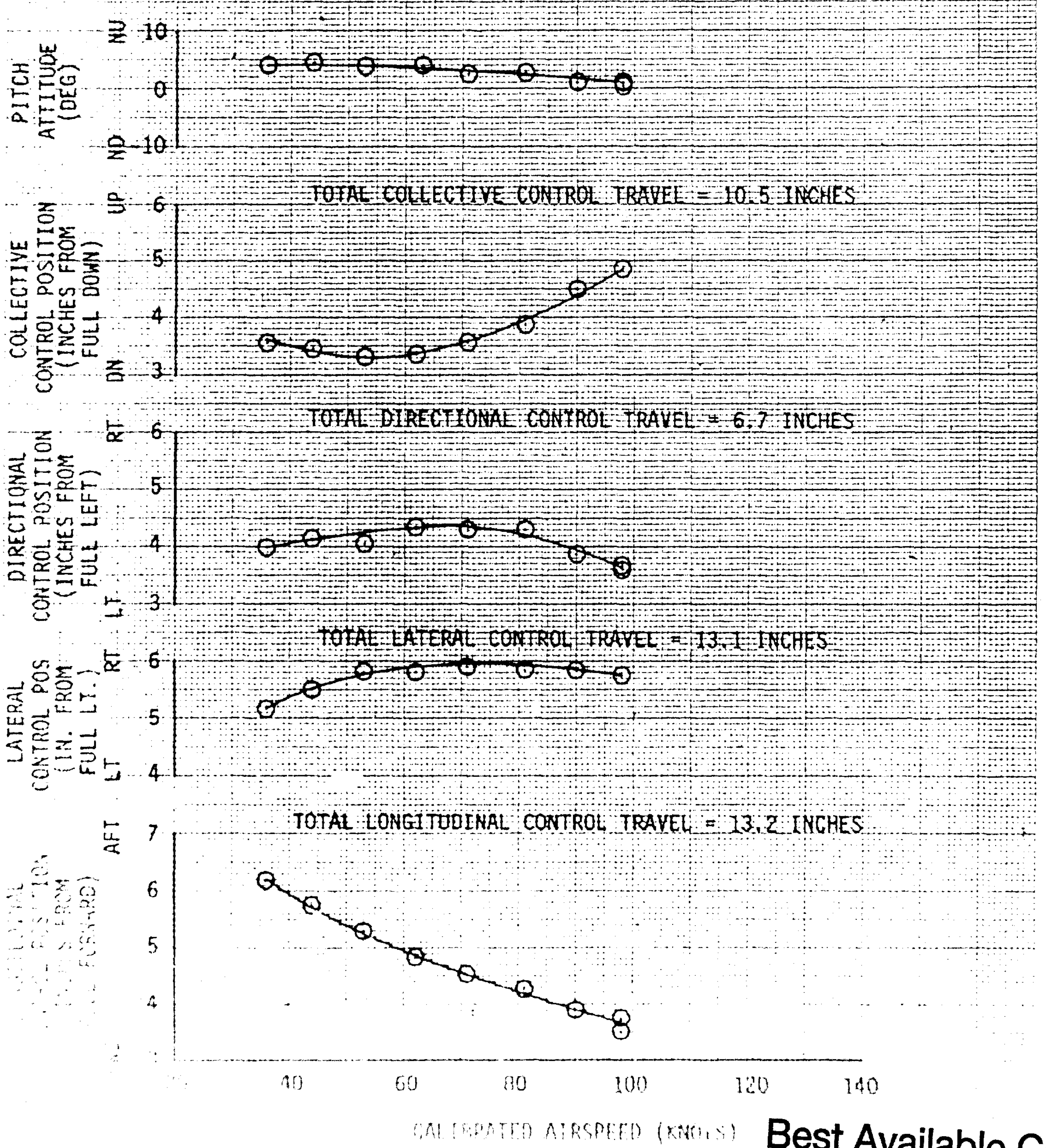

FIGURE 5
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID FLIGHT CONDITION	PBDS CONFIGURATION
○	8140	135.8 MID	1.2	8020	2.5	324	LEVEL	DEFLATED
□	7880	134.6 MID	1.2	9160	0.5	324	LEVEL	VENTED

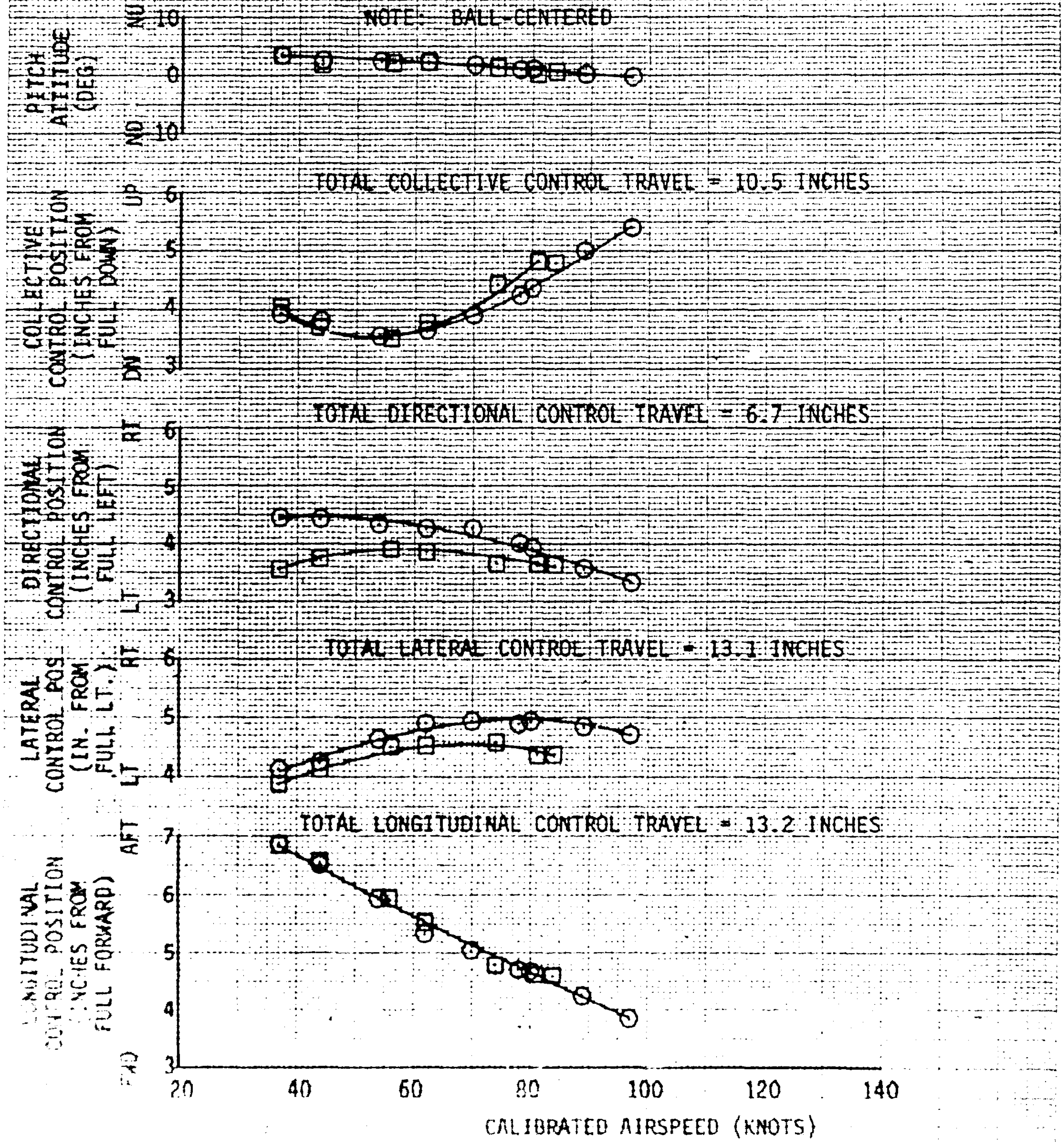

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FIGURE 6
SIDEWARD FLIGHT
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT (FT)	PBDS CONFIGURATION
7890	138.0(MID)	-0.5	3220	20.5	324	10	DEFLATED

NOTE: T DENOTES EXTREME TRAVEL FROM TRIM DURING ATTEMPTED STABILIZED POINT

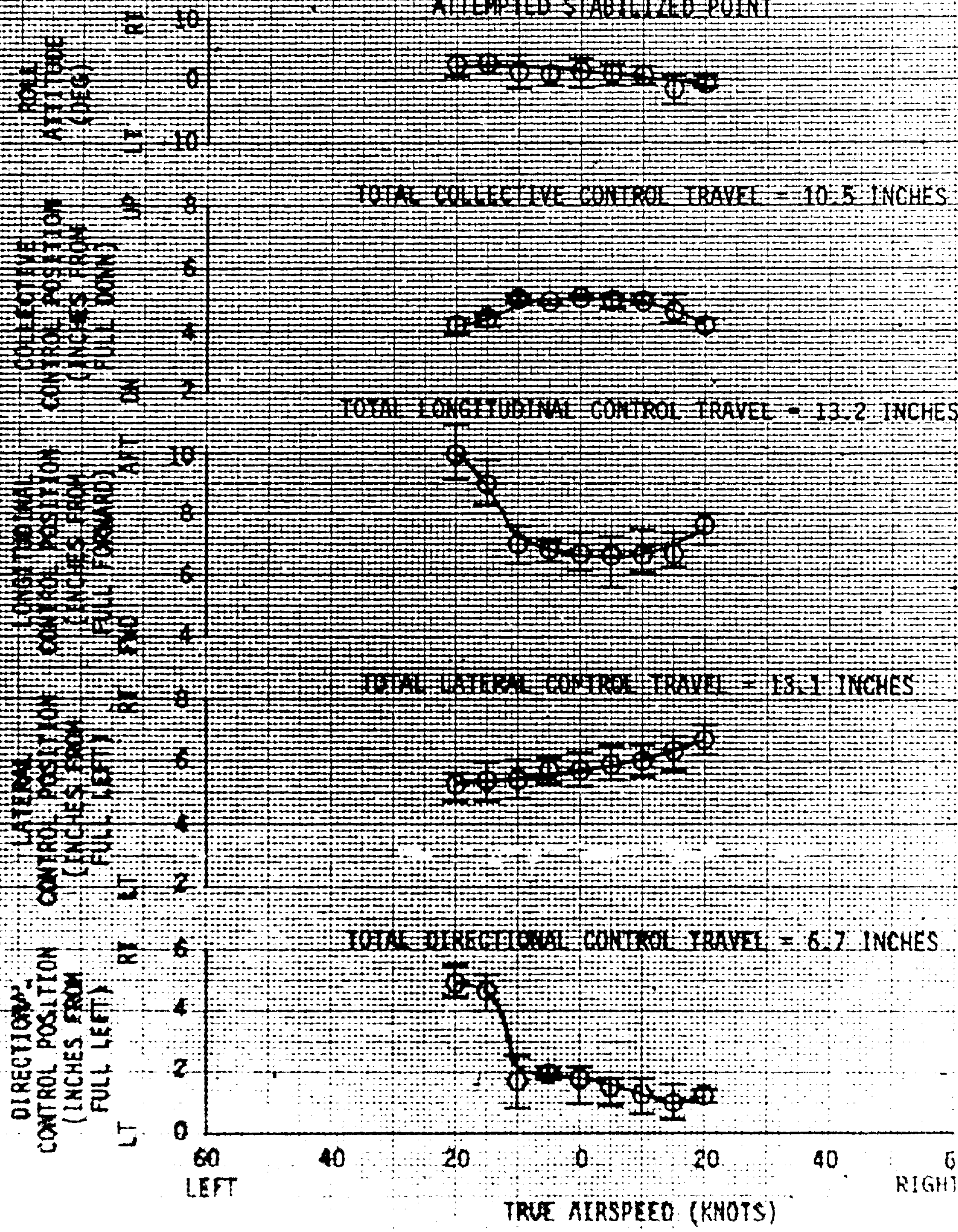

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FIGURE 7
SIDEWARD FLIGHT
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT (FT)	PBDS CONFIGURATION
7640	137.2 MID	0.5	3220	20.5	324	10	VENTED

NOTE: I DENOTES EXTREME TRAVEL FROM TRIM DURING ATTEMPTED STABILIZED POINT

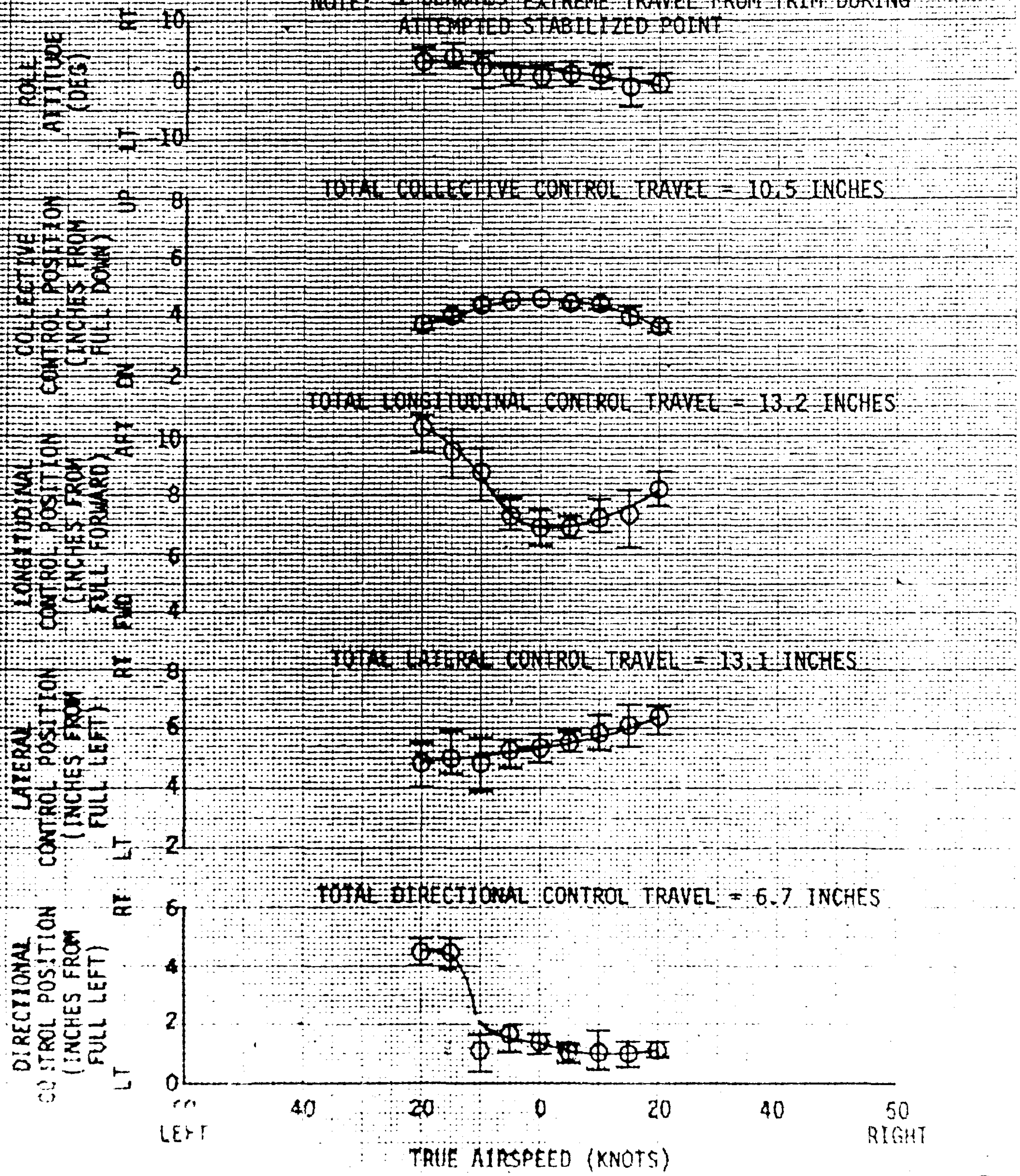

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FIGURE 8
LOW SPEED FORWARD AND REARWARD FLIGHT
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT (FT)	PBDS CONFIGURATION
7900	138.3	MID 0.6	3220	20.5	324	10	DEFLATED

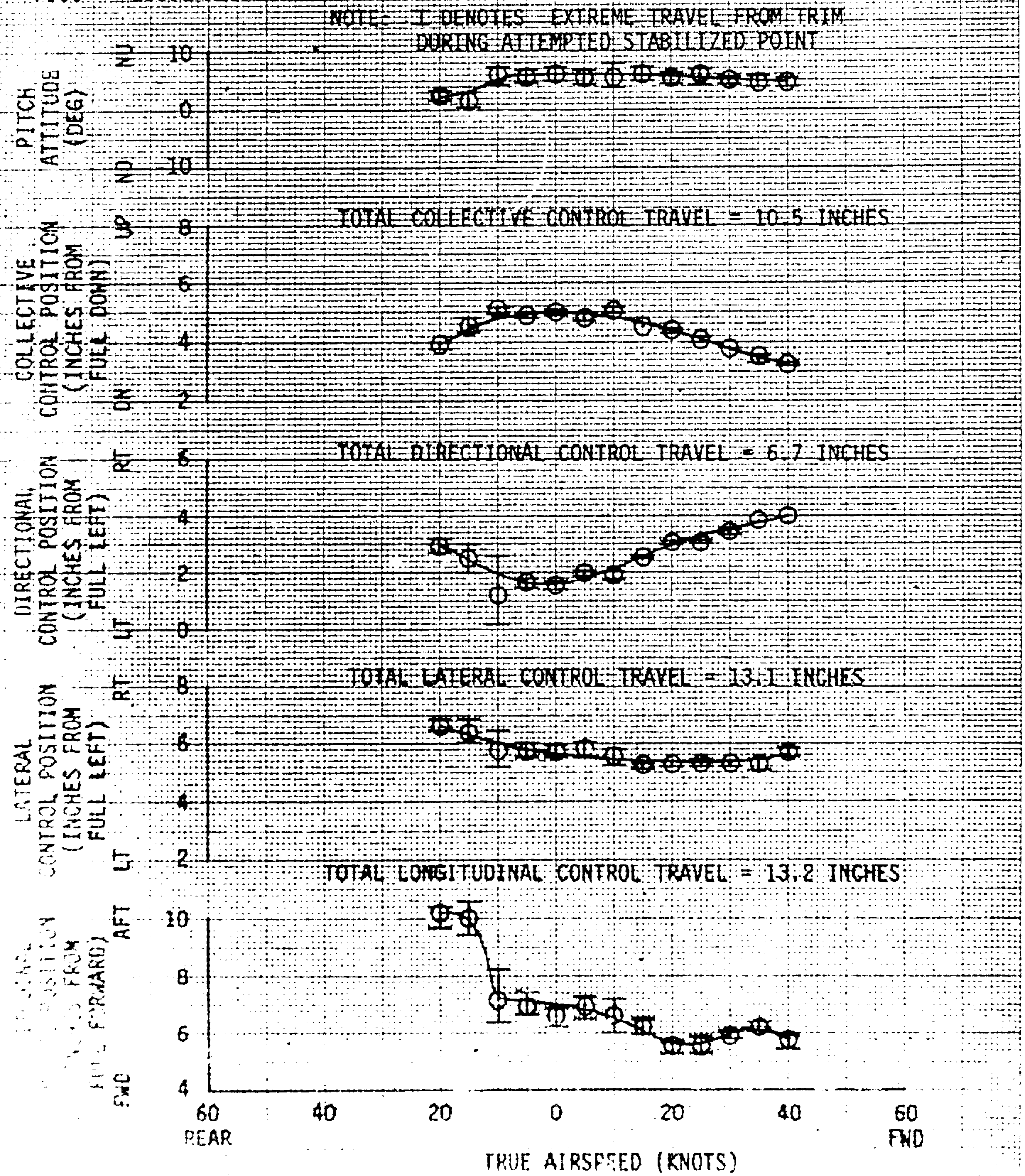

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FIGURE 9
LOW SPEED FORWARD AND REARWARD FLIGHT
JUH-1H USA S/N 70-16318
PNEUMATIC BOOT DEICE SYSTEM

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FEET)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	SKID HEIGHT (FT)	RBDS CONFIGURATION
7680	137.3 MID	0.5	3220	20.5	324	10	VENTED

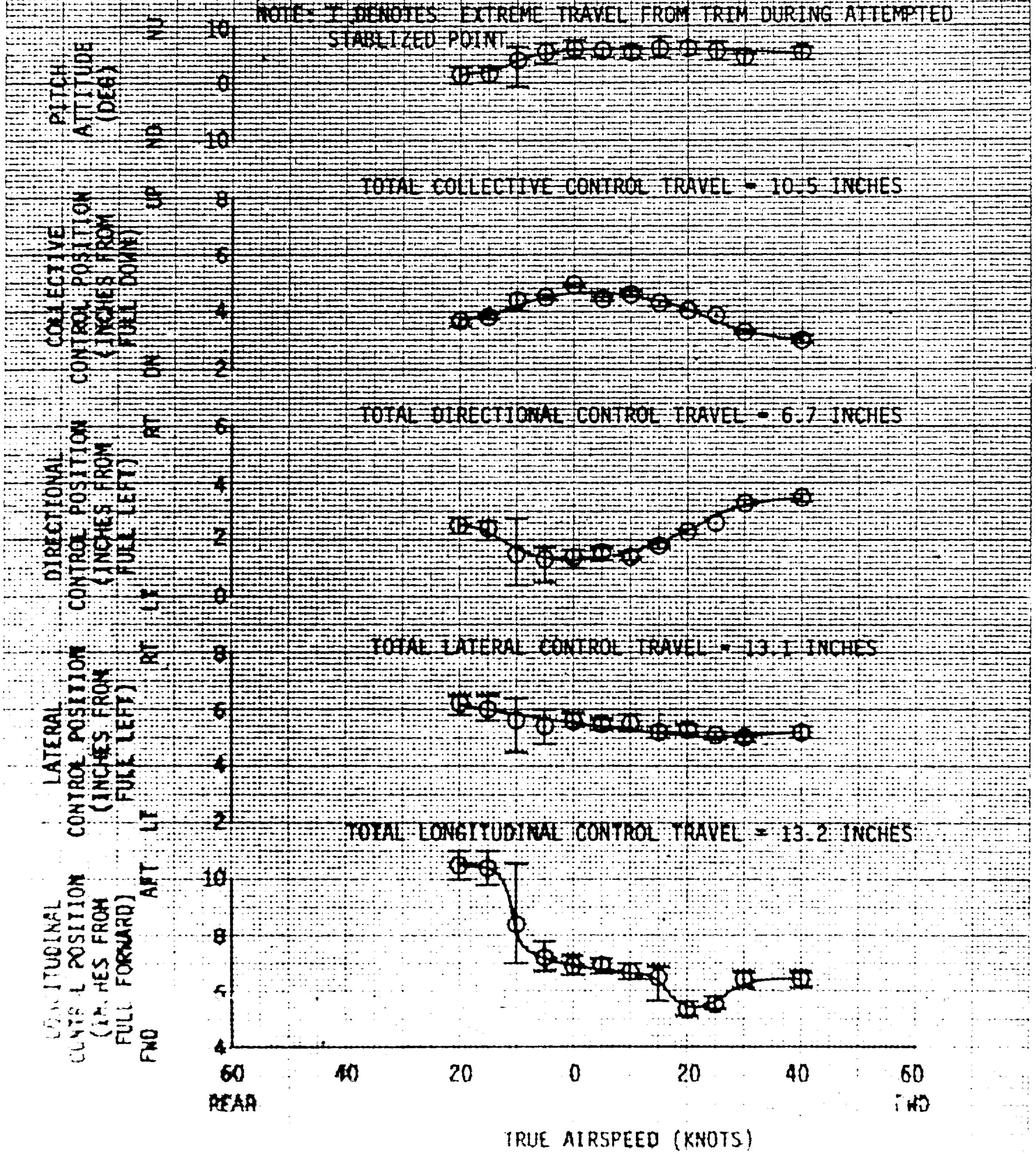
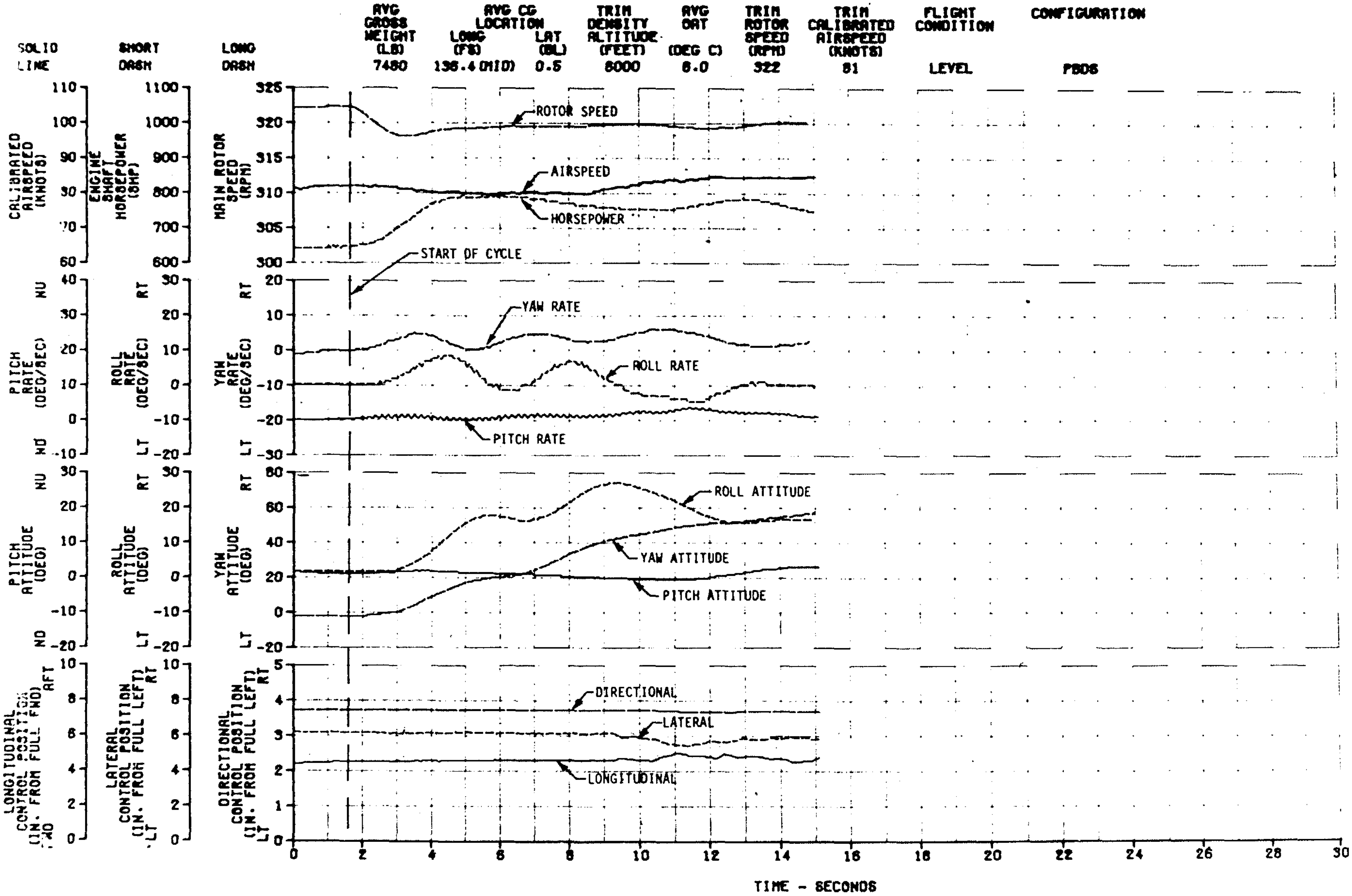


FIGURE 10

CONTROL-FIXED PBOS ACTIVATION CYCLE

JUH-1H USA S/N 70-16318



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APPENDIX F. UH-1H CONTROL SYSTEM LOADS MONITORING

1. To simplify real-time monitoring of the control system loads, only pitch link loads were monitored. Bell Helicopter Textron (Mr. J.A. White) provided the following information concerning the "trail" of the pitch link loads through the UH-1H control system.

2. The load at each location in the control system is presented in terms of a "coefficient" times the load at the pitch link. Schematic diagrams of the control system are presented in figures 1 and 2 to aid in the location and identification of a particular load in the control stream. The loads are grouped in terms of rotating system components (table 1) and fixed components (table 2) since this is the natural point of separation. Monitored ground and flight loads are presented in tables 3 through 8.

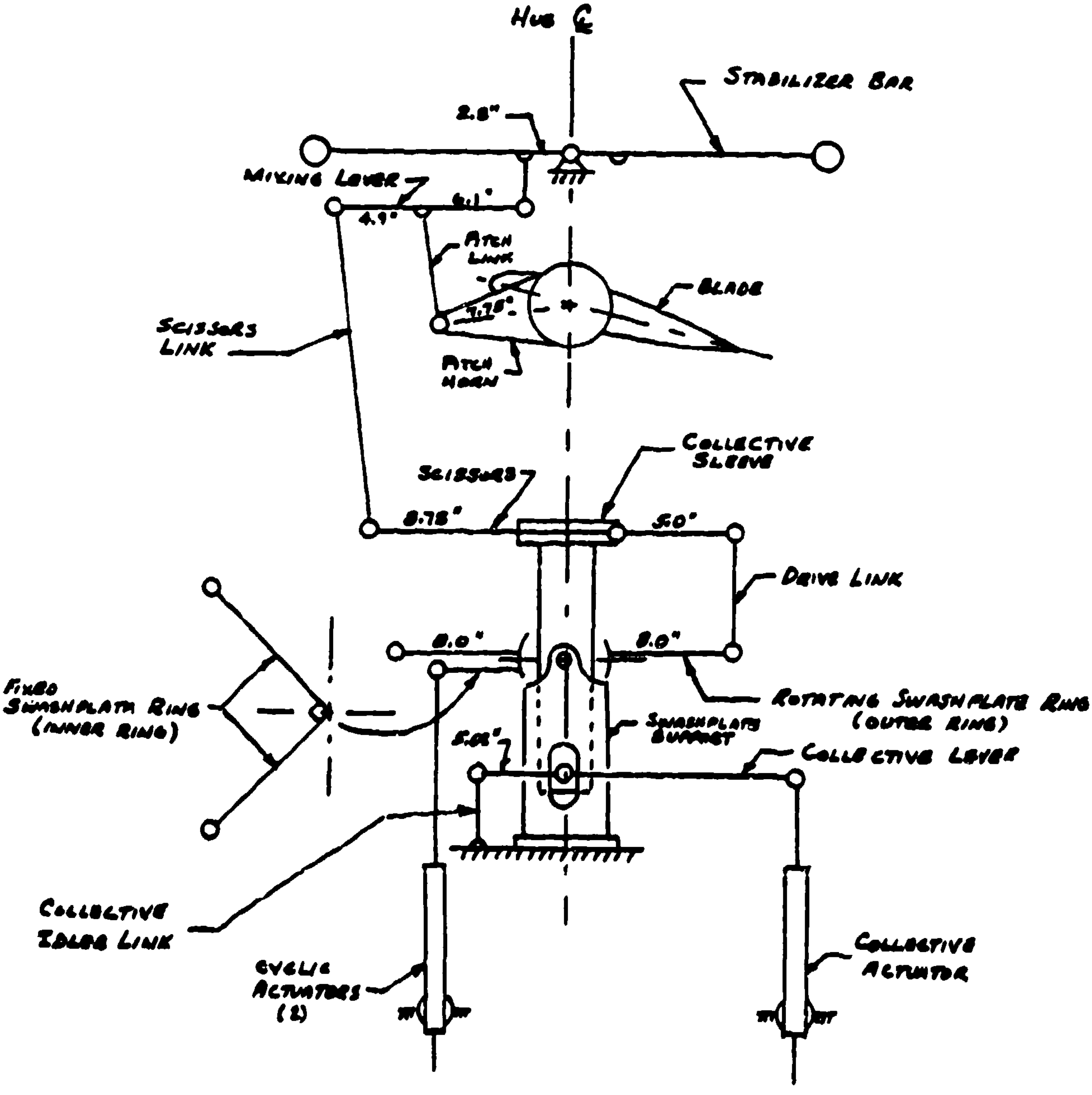


Figure 1. Model UH-1H Control System Schematic

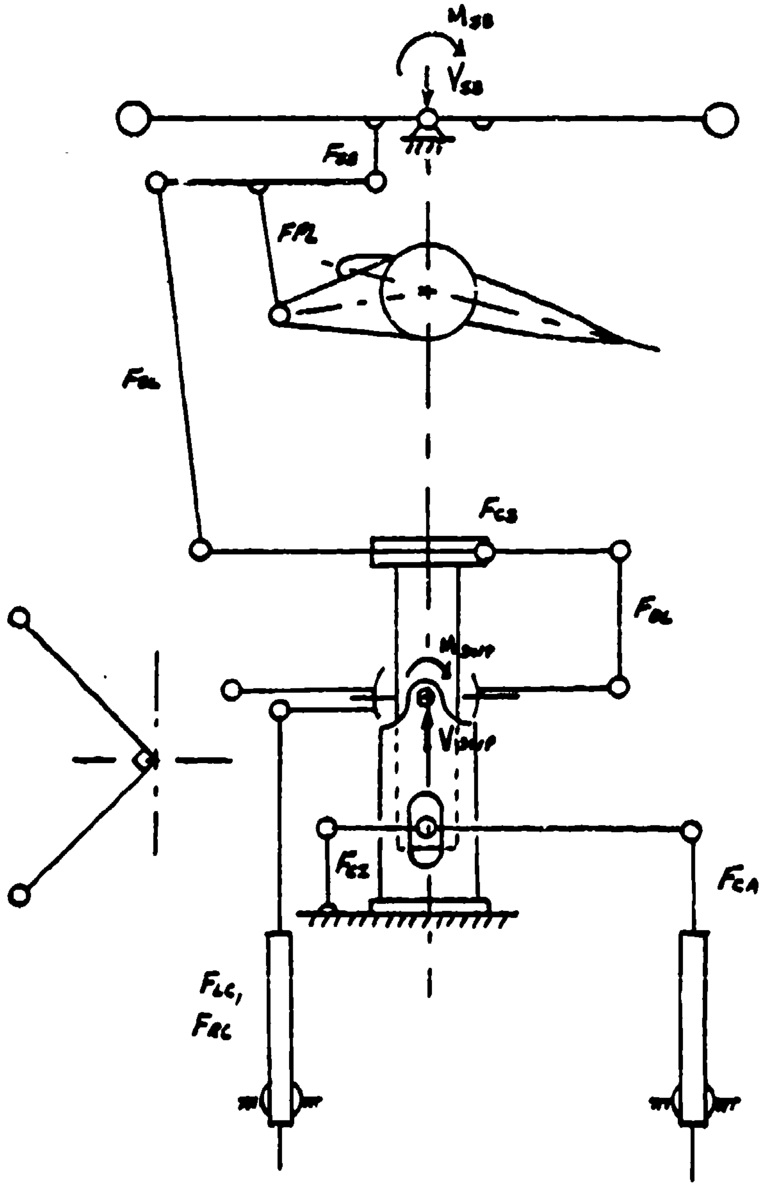
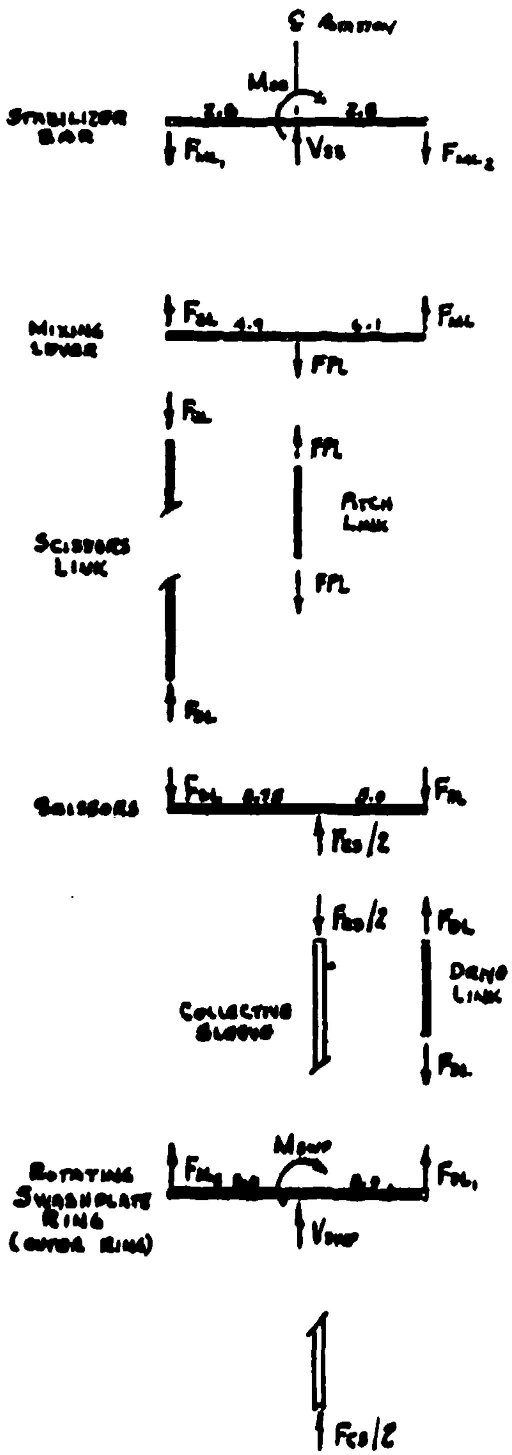


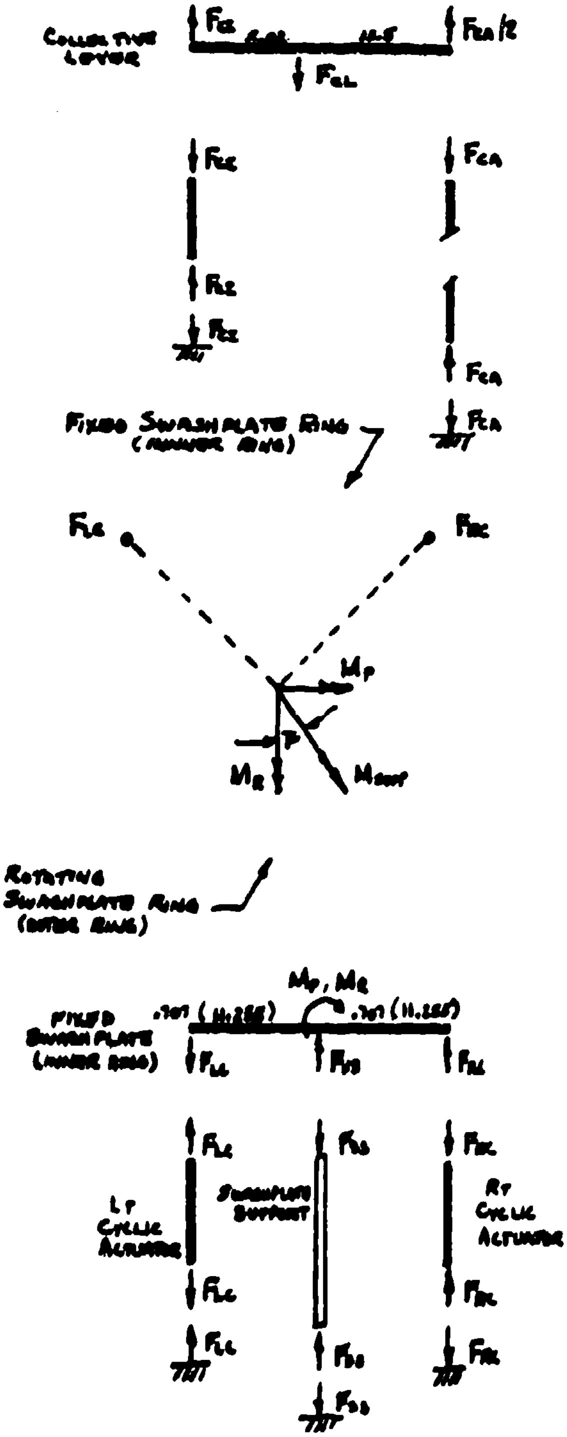
Figure 2. Model UH-1H Control Load Schematic

Table 1. Model UH-1H Control System Loads (Rotating System)



Load Coefficient	Nomenclature
V_{SB} = .8708 FPL	Stabilizer Bar Vertical Shear (Collective Loading)
M_{SB} = 2.4942 FPL	Stabilizer Bar Moment (Cyclic Loading)
F_{ML} = .4454 FPL	Mixing Lever Vertical Load
F_{SL} = .5546 FPL	Scissors Link Axial Load
FPL = Reference Load	Pitch Link Axial Load $FPL = F_{PL_0} + \sum_{i=1}^{n} F_{PL_i} \cos n(\psi - \phi_i)$
F_{DL} = .9708 FPL	Drive Link Axial Load
F_{CS} = 1.5252 FPL/Bld	Collective Sleeve Axial Load/Bld (Rotating Sys)
M_{SWP} = 15.8296 FPL	Swashplate Rotating Moment (Cyclic Loading)
V_{SWP} = 1.9412 FPL	Swashplate Rotating Shear (Collective Loading)

Table 2. Model UH-1H Control System Loads
(Fixed System)



Load Coefficient	Nomenclature
$F_{CI} = \underline{1.1118}$ FPL	Collective Idler Axial Load (Rt & Lt Leg)
$F_{CA} = \underline{.8268}$ FPL	Collective Actuator Axial Load
$M_P = \underline{15.5296 \sin\psi}$ FPL	Swashplate Pitching Moment
$M_R = \underline{15.5296 \cos\psi}$ FPL	Swashplate Rolling Moment
$F_{RC} = \underline{1.952\left[\cos n\psi(\sin\psi + \cos\psi)\right]}\ FPL_n$ for $n = 1,3,5,\ldots$	Rt Cyclic Boost Tube Axial Load
$F_{LC} = \underline{1.952\left[\cos n\psi(\sin\psi - \cos\psi)\right]}\ FPL_n$ for $n = 1,3,5,\ldots$	Lt Cyclic Boost Tube Axial Load
$F_{SS} = \underline{-.976\,(\cos n\psi \cos\psi)}\ FPL_n$	Swashplate Support Axial Load

Table 3. Monitored Flight Loads

Flight Condition	Main Rotor Beam Bend. Moment (Blade Station 35) in.-lb						Main Rotor Blade Chord Bend. Moment (Blade Station 35) 1000 in.-lb						Main Rotor Blade Beam Bend. Moment (Blade Station 84) in.-lb					
	Boot Condition				System Vented		Boot Condition				System Vented		Boot Condition				System Vented	
	Normal		Cycled				Normal		Cycled				Normal		Cycled			
	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[3]	Stdy	Osc[3]	Stdy	Osc[3]
	NOTE 4																	
Stabilized Level Flight,																		
50 KIAS	4921	6625	8707	6625	8707	5205	104.3	29.1	118.2	41.6	109.9	30.5	10,025	2467	10,025	2402	9445	2612
60 KIAS	4921	6625	5867	6625	5867	6625	107.1	22.2	121.0	30.5	107.1	25.0	9735	2612	10,315	2557	10,025	2759
70 KIAS	5867	6625	12,492	6517	6814	7571	104.3	26.4	123.9	33.3	110.0	25.0	9735	2902	10,896	3483	10,315	3423
80 KIAS	4921	8044	11,546	10,410	8706	12,303	109.9	29.1	134.8	44.4	109.9	30.5	10,023	3628	10,606	4063	10,315	4353
85 KIAS	6814	8517	16,278	11,356	10,599	9464	112.6	29.1	118.2	45.8	121.0	37.5	10,896	3918	12,057	5950	11,186	4063
90 KIAS	8707	9464	14,385	12,303	10,599	13,249	112.6	34.7	129.3	50.0	126.5	41.6	10,606	4208	11,766	5804	10,896	5369
100 KIAS	8797	11,356	20,065	18,927	20,063	15,142	121.0	36.1	126.5	69.4	134.9	60.0	11,186	4644	13,798	6820	12,927	5950
60 KIAS Climb @ 35 psi Q[5]	6814	8517	8707	8990	7760	10,410	107.1	27.8	143.2	34.7	104.3	38.9	10,606	3483	10,606	3628	10,606	3918
60 KIAS Descent @ 14 psi Q	-757	6625	1136	7571	-1704	5678	90.4	22.2	107.1	34.7	104.3	27.8	8284	2322	8574	2322	8574	2177
60 KIAS Climb @ 43 psi Q	12,492	10,410	—	—	17,224	13,722	132.1	30.5	—	—	134.9	51.4	11,766	3773	—	—	12,637	4353
90 KIAS Climb @ 39 psi Q	8707	11,356	14,385	15,142	10,599	12,303	118.2	31.9	137.6	70.8	132.1	41.6	11,476	4353	12,637	5514	12,347	3338
90 KIAS Descent @ 24 psi Q	1136	8044	6814	8991	3028	7571	107.1	33.3	132.1	41.6	112.6	33.3	10,025	4063	10,315	4353	10,606	4353
15° Banked Turn to left	3975	8517	10,599	12,776	16,599	11,356	112.6	29.1	134.8	59.7	132.1	52.7	10,315	4062	10,606	5369	10,606	5514
30° Banked Turn to left	15,331	10,410	19,117	12,776	14,385	15,142	121.0	50.0	132.1	61.1	132.1	65.2	12,347	4644	12,637	5659	12,927	6820
15° Banked Turn to right	7760	10,883	11,546	12,776	7760	10,410	112.6	31.9	134.8	61.1	129.3	38.9	10,606	4353	11,186	5224	11,186	4934
30° Banked Turn to right	8707	10,410	11,420	15,142	21,956	12,303	115.4	51.4	126.5	72.2	126.5	55.5	12,347	4644	14,378	7546	12,927	6385
	Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented	
	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc
70 KIAS Autorotation[6]	3028[7] -5480	6625[7] 7571	3975	5678	-4533	8517	109.9[7] 65.4	27.8[7] 31.9	71.0	34.7	68.2	27.8	9154[7] 8284	3438[7] 3047	9445	4494	7703	3338
80 KIAS Autorotation[6]	2082[7] -6435	6625[7] 7571	-757	10,410	-4543	5205	109.9[7] 71.0	18.0[7] 33.3	71.0	36.1	57.1	26.4	10,025 7413	3483[7] 3483	10,025	4789	8284	3628
90 KIAS Autorotation[6]	4921[7] -7382	10,410[7] 8044	3028	14,196	-6435	6151	115.4[7] 65.4	31.9[7] 36.1	71.0	41.6	65.5	30.5	10,315 4703	4353[7] 3918	8864	5224	7994	3773

NOTES:

[1]Oscillatory endurance limit ± 27,500 in.-lb
[2]Oscillatory endurance limit ± 90,000 in.-lb
[3]Oscillatory endurance limit ± 8800 in.-lb
[4]Stdy = Steady, Osc = Oscillatory
[5]Q= torque
[6]Boot not cycled during autorotation
[7]At entry airspeed

Table 4. Monitored Flight Loads

Flight Condition	Main Rotor Chord Bend. Moment (Blade Station 84) 1000 in.-lb						Main Rotor Mast Parallel Bend. Moment (Blade Station 23) 1000 in.-lb						Main Rotor Mast Torque 1000 in.-lb					
	Boot Condition				System Vented		Boot Condition				System Vented		Boot Condition				System Vented	
	Normal		Cycled				Normal		Cycled				Normal		Cycled			
	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc	Stdy	Osc	Stdy	Osc
	NOTE 3																	
Stabilized Level Flight, 50 KIAS	75.8	21.1	82.1	29.6	77.9	22.2	0	5678	0	7571	0	6309	95.6	1.6	133.4	3.3	118.6	2.5
60 KIAS	73.7	16.9	83.2	23.2	80.2	19.9	0	5047	0	6625	0	5678	97.3	1.6	139.9	3.3	118.6	1.6
70 KIAS	75.8	21.1	80.0	29.6	80.0	23.2	0	5994	0	7571	0	5994	100.6	3.3	146.5	2.5	133.5	2.5
80 KIAS	77.9	21.1	86.4	37.0	82.1	21.1	0	6309	0	8833	0	6940	118.6	2.5	167.8	4.1	131.7	3.3
85 KIAS	80.0	19.0	86.4	35.9	84.9	27.5	0	6309	0	9148	0	7591	118.6	1.6	164.5	3.3	144.9	2.5
90 KIAS	80.0	23.3	92.7	39.1	86.4	20.5	0	6625	0	9464	0	7886	128.4	3.3	171.1	4.1	144.9	3.3
100 KIAS	86.4	23.2	96.9	54.0	94.8	42.3	0	8833	0	11,356	0	10,726	148.1	2.5	189.1	29.5	180.9	3.3
60 KIAS Climb @ 35 psi Q[4]	84.9	14.8	94.8	31.7	86.4	26.4	0	7886	0	8833	0	8202	136.6	1.6	187.5	2.5	158.0	2.5
50 KIAS Descent @ 14 psi Q	65.2	15.8	76.0	20.1	73.7	15.9	0	4732	0	6309	0	5047	67.8	1.6	100.6	1.6	82.5	2.5
60 KIAS Climb @ 43 psi Q	92.7	22.2	—	—	92.7	40.2	0	9464	0	—	0	11,041	174.4	2.5	—	—	192.4	23.1
90 KIAS Climb @ 39 psi Q	88.5	21.1	107.5	45.4	94.8	65.5	0	8517	0	11,356	0	9464	151.4	1.6	189.1	34.4	177.7	1.6
90 KIAS Descent @ 24 psi Q	77.9	23.2	86.4	27.5	82.1	22.2	0	5363	0	7886	0	6625	95.6	1.6	141.6	2.5	120.2	2.5
15° Banked Turn to left	82.1	19.0	94.8	44.4	92.7	38.8	0	6625	0	9779	0	8833	125.2	2.5	172.7	3.3	167.8	3.3
30° Banked Turn to left	88.5	29.6	94.8	40.2	96.9	46.5	0	8202	0	10,095	0	8833	146.5	3.3	180.9	6.6	176.0	2.5
15° Banked Turn to right	80.0	23.2	94.8	42.3	94.8	31.7	0	6940	0	9779	0	8202	130.1	1.6	179.3	3.3	156.3	4.1
30° Banked Turn to right	82.1	30.6	94.8	55.0	94.8	46.5	0	8517	0	10,410	0	9464	131.7	1.6	185.9	4.1	169.5	4.9
	Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented	
	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc
70 KIAS Autorotation[5]	75.8[6] 52.5	16.9[6] 21.1	52.5	25.4	46.2	18.9	0	6940[6] 6940	0	14,511	0	5994	108.8[6] 6.0	1.6[6] .8	6.0	27.9	6.0	.8
80 KIAS Autorotation[5]	75.8[6] 48.3	18.0[6] 24.3	46.2	26.4	44.1	21.1	0	6309[6] 7256	0	7256	0	5678	110.4[6] 6.0	1.6[6] .8	6.0	29.5	6.0	.8
90 KIAS Autorotation[5]	84.2[6] 48.3	16.9[6] 25.4	46.2	27.5	46.2	23.2	0	6625[6] 6309	0	6940	0	5994	131.7[6] 6.0	3.3[6] .8	6.0	32.8	6.0	.8

NOTES:

[1]Oscillatory endurance limit ± 60,000 in.-lb
[2]Oscillatory endurance limit ± 34,000 in.-lb
[3]Stdy = Steady, Osc = Oscillatory
[4]Q= torque
[5]Boot not cycled during autorotation
[6]At entry airspeed

Table 5. Monitored Flight Loads

Flight Condition	Main Rotor Yoke Chord Bend Moment (Blade Station 6.3) 1000 in.-lb						Main Rotor Blade Beam Bend Moment (Blade Station 192) 1000 in.-lb						Main Rotor Blade Chord Bend. Moment (Blade Station 192) 1000 in.-lb					
	Boot Condition				System Vented		Boot Condition				System Vented		Boot Condition				System Vented	
	Normal		Cycled				Normal		Cycled				Normal		Cycled			
	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[3]	Stdy	Osc[3]	Stdy	Osc[3]
	NOTE 4																	
Stabilized Level Flight, 50 KIAS	40	30	60	42.5	55	32.5	-5.0	2.7	-4.3	2.5	-4.5	2.5	-9.5	5.0	-6.0	7.0	-5.0	5.0
60 KIAS	40	25	55	37.5	55	37.5	-5.0	3.0	-4.5	3.0	-4.8	3.0	-9.0	4.5	-5.0	6.0	-5.0	5.0
70 KIAS	40	27.5	60	49	50	40	-5.0	3.0	-4.3	3.25	-4.3	3.0	-9.0	5.0	-5.0	6.0	-5.0	5.0
80 KIAS	55	30	85	62.5	60	37.5	-5.0	3.0	-4.0	4.0	-5.0	3.2	-5.0	5.0	-2.0	10.0	-5.0	5.0
85 KIAS	50	25	70	65	70	50	-5.0	3.05	-4.0	4.0	-4.2	3.8	-7.5	3.0	-7.5	7.5	-5.0	6.5
90 KIAS	51	32.5	75	60	67.5	45	-4.8	3.9	-4.0	4.25	-3.75	3.9	-7.5	5.0	-5.0	8.0	-5.0	5.0
100 KIAS	75	40	85	90	75	75	-4.0	3.8	-5.0	5.0	-4.2	4.5	-5.0	5.0	-5.0	14.0	-5.0	10.0
60 KIAS Climb @ 35 psi Q[5]	60	27	80	50	70	45	-4.05	2.8	-4.0	3.1	-4.0	3.0	-6.0	5.0	-2.0	7.0	-5.0	6.0
60 KIAS Descent @ 14 psi Q	25	25	30	25	30	25	-5.5	2.0	-5.0	2.4	-5.0	2.4	-10.0	4.5	-9.0	4.0	-8.5	4.0
60 KIAS Climb @ 43 psi Q	75	30	—	—	85	65	-3.8	3.0	—	—	-3.9	3.5	-3.5	4.0	—	—	-10.0	10.0
90 KIAS Climb @ 39 psi Q	75	35	120	75	80	60	-4.1	3.5	-3.8	4.5	-4.0	4.0	-6.0	3.5	0	10.0	-3.0	8.0
90 KIAS Descent @ 24 psi Q	40	30	65	45	55	40	-5.0	3.2	-4.0	4.0	-4.2	3.8	-8.0	4.0	-5.0	5.5	-6.0	4.0
15° Banked Turn to left	60	30	70	70	80	60	-4.5	3.4	-4.0	4.0	-4.0	4.0	-6.0	3.75	-2.0	10.0	-3.0	8.0
30° Banked Turn to left	70	50	80	70	80	70	-5.0	4.0	-4.0	4.0	-4.5	4.5	-5.0	5.0	-4.0	10.0	-2.0	10.0
15° Banked Turn to right	60	32.5	75	62.5	75	50	-4.0	3.6	-4.0	4.5	-4.0	4.0	-6.0	4.0	-5.0	10.0	-5.0	6.0
30° Banked Turn to right	60	40	80	87.5	75	75	-4.2	3.9	-5.0	5.0	-4.5	4.7	-6.0	5.0	0	13.5	-3.0	10.0
	Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented	
	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc
70 KIAS Autorotation[6]	40[7] -15	22[7] 30	-10	37.5	-15	30	-5.0[7] -6.0	3.0[7] 2.5	-6.0	3.5	-6.0	2.5	7.5[7] 7.5	4.5[7] 4.0	-12.5	5.0	-7.5	4.0
80 KIAS Autorotation[6]	50[7] -15	25[7] 30	0	35	-10	25	-5.0[7] -6.0	3.0[7] 2.7	-5.8	3.5	-5.8	3.0	7.5[7] -14.0	2.5[7] 4.5	-12.5	7.0	-14.0	4.0
90 KIAS Autorotation[6]	60[7] -20	30[7] 35	-15	40	-10	25	-4.5[7] -5.8	3.5[7] 3.0	-5.8	4.5	-5.8	3.5	-6.0[7] -12.0	3.0[7] 5.0	-14.0	6.5	-14.0	4.5

NOTES:

[1]Oscillatory endurance limit ± 145,000 in.-lb
[2]Oscillatory endurance limit ± 7500 in.-lb
[3]Oscillatory endurance limit ± 20,000 in.-lb
[4]Stdy = Steady, Osc = Oscillatory
[5]Q= torque
[6]*Boot not cycled during autorotation*
[7]At entry airspeed

Table 6. Summary of Measured Loads

Flight Condition	Red Pitch Link Axial Load (lb)						Main Rotor Mast Perpendicular Moment 1000 in.-lb						Main Rotor Yoke Beam Bend Moment (Blade Station 6.3) 1000 in.-lb					
	Boot Condition				System Vented		Boot Condition				System Vented		Boot Condition				System Vented	
	Normal		Cycled				Normal		Cycled				Normal		Cycled			
	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[1]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[2]	Stdy	Osc[3]	Stdy	Osc[3]	Stdy	Osc[3]
	NOTE 4																	
Stabilized Level Flight, 50 KIAS	240	275	0	350	200	350	0	5.25	0	8.0	0	5.5	-25	20	-20	25	-15	20
60 KIAS	250	275	0	450	20	400	0	5.1	0	6.5	0	6.0	-22	20	-15	25	-20	20
70 KIAS	250	350	0	500	100	375	0	5.5	0	9.0	0	6.0	-20	20	-20	30	-20	20
80 KIAS	275	475	0	750	200	500	0	6.5	0	10.0	0	6.5	-28	20	-15	35	-20	21
85 KIAS	250	500	50	650	50	525	0	6.0	0	9.5	0	7.5	-30	15	-20	30	-18	15
90 KIAS	250	600	150	725	125	550	0	7.5	0	10.0	0	8.0	-20	20	-10	31	-20	20
100 KIAS	260	600	50	800	200	700	0	9.0	0	12.0	0	11.0	-20	22	0	42	-10	27
60 KIAS Climb @ 35 psi Q[5]	350	375	50	600	200	400	0	8.0	0	10.0	0	8.5	-30	15	-17	23	-30	15
60 KIAS Descent @ 14 psi Q	200	300	0	350	0	350	0	5.0	0	6.0	0	5.0	-37	20	-30	32.5	-32	15
60 KIAS Climb @ 43 psi Q	350	400	—	—	200	375	0	10.0	0	—	0	10.0	-18	11	—	—	-20	15
90 KIAS Climb @ 39 psi Q	300	550	200	700	200	600	0	9.0	0	11.0	0	10.0	-28	19	-10	-10	-22	20
90 KIAS Descent @ 24 psi Q	250	500	150	600	100	500	0	5.0	0	8.0	0	6.5	-35	20	-30	-30	-30	15
15° Banked Turn to left	350	500	100	600	200	650	0	7.0	0	10.0	0	9.0	-30	[illegible]	-30	-30	-18	25
30° Banked Turn to left	250	600	50	650	200	800	0	8.5	0	10.0	0	10.0	0	[illegible]	-20	-20	20	30
15° Banked Turn to right	300	500	50	650	100	500	0	6.5	0	10.0	0	7.5	-20	19	-20	-20	-20	19
30° Banked Turn to right	300	550	200	1000	250	800	0	7.5	0	11.5	0	10.0	-20	19	-20	10	10	30
	Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented		Normal		Throttle Chop		System Vented	
	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc	Stdy	Osc
70 KIAS Autorotation[6]	300[7] -200	375[7] 400	-100	350	-100	350	0 0	5.5[7] 7.5	0	7.5	0	7.5	-40[7] -38	15[7] 20	-40	20	-38	20
80 KIAS Autorotation[6]	300[7] -100	450[7] 375	-100	375	-250	350	0 0	6.0[7] 6.0	0	6.0	0	6.0	-36[7] -40	15[7] 18	-40	19	-30	15
90 KIAS Autorotation[6]	350[7] -100	350[7] 350	-150	450	-200	375	0 0	7.0[7] 6.0	0	6.0	0	6.0	-38 -40	20[7] 80	-40	19	-25	15

NOTES:

[1]Oscillatory endurance limit ± 800 lb
[2]Oscillatory endurance limit ± 37,000 in.-lb
[3]Oscillatory endurance limit ± 145,000 in.-lb
[4]Stdy = Steady, Osc = Oscillatory
[5]Q = torque
[6]Boot not cycled during autorotation
[7]At entry airspeed

Table 7. Monitored Ground Loads Survey[1]

Condition Engine Torque Setting		324 RPM								294 RPM							
		Cycle			Sustained				Off	Cycle			Sustained				Off
		Min	Max	Δ	Min	Max	Δ	Steady	Starts/Stop Δ	Min	Max	Δ	Min	Max	Δ	Steady	Starts/Stop Δ
		52800 271.4	63200 324.8	10400 53.4	52800 271.4	60600 311.5	7800 40.1	52800 271.4	0 0	42400 197.6	47600 221.8	5200 24.2	45000 209.7	50200 233.9	5200 24.2	45000 209.7	2600 12.1
15 psi		68400 351.6	78800 405.0	10400 53.4	71000 364.9	78800 405.0	78800 40.1	71000 364.9	2600 13.4	65800 301.6	73600 34300	7800 36.3	71000 330.9	76200 355.1	5200 24.2	68400 318.7	2600 12.1
20 psi		86600 445.1	97000 498.6	10400 53.4	89200 458.5	94400 485.2	3200 16.4	89200 485.4	2600 13.4	84000 391.4	94000 439.9	10400 48.5	86600 403.6	94400 439.9	7800 36.3	86600 403.6	2600 12.1
25 psi		107400 552.0	120400 618.8	13000 66.8	110000 565.4	120400 618.8	10400 53.4	110000 565.4	2600 13.4	107400 500.5	117800 549.0	10400 48.5	107400 500.5	115200 536.8	7800 36.3	107400 500.5	0 0
30 psi	in-lb s'p	125607 645.6	136000 699.0	10400 53.4	128200 658.9	138600 712.4	10400 53.4	125600 645.6	0 0	120400 561.1	133400 6621.6	13000 60.6	123000 573.2	133400 621.6	10400 48.5	123000 573.2	2600 12.1
35 psi		141200 725.8	156800 806.0	15600 80.2	146400 752.8	156800 806.0	10400 53.4	143800 739.1	2600 13.4	138600 645.9	151600 706.4	13000 60.6	141200 658.0	154200 718.6	12000 60.6	163800 763.3	3200 14.9
40 psi		164600 846.0	182800 939.6	18200 93.5	167200 859.4	180200 926.2	13000 66.8	167200 859.4	2600 13.4	162000 754.9	177600 827.6	15600 72.7	164600 767.0	180200 839.7	15600 72.7	167200 779.2	3200 14.9
45 psi		182800 939.6	201000 1033.1	18200 935	188000 966.3	203600 1046.5	15600 80.2	188000 966.8	3200 16.4	182800 85186	198400 924.5	15600 72.7	185400 864.0	198400 924.5	13000 60.6	198000 922.7	3200 14.9

NOTE:

[1]Aircraft ballasted to 9500 lbs gross weight and tethered to the ground

Table 8. Main Rotor Horsepower Trends Monitored During Flight Loads Survey Testing[1]

Flight Condition	Boot Normal	Boot Cycled	Δ Horsepower	System Vented	Δ Horsepower
Stabilized Level Flight	492	686	194	610	118
50 KIAS	500	719	219	610	110
60 KIAS	517	753	236	635	118
70 KIAS	610	863	253	677	67
80 KIAS	610	846	236	745	135
90 KIAS	660	880	220	745	85
100 KIAS	762	972	210	930	168
60 KIAS Climb @ 35 psi Q	702	964	262	812	110
60 KIAS Descent @ 14 psi Q	323	517	194	424	101
60 KIAS Climb @ 43 psi Q	896	--	--	989	93
90 KIAS Climb @ 39 psi Q	778	972	194	913	135
90 KIAS Descent @ 24 psi Q	492	728	236	618	126
70 KIAS Autorotation[2]	--	--	--	--	--
80 KIAS Autorotation	--	--	--	--	--
90 KIAS Autorotation	--	--	--	--	--
15° Banked Turn to Left	643	888	245	863	220
30° Banked Turn to Left	753	930	177	905	152
15° Banked Turn to Right	669	922	253	804	135
30° Banked Turn to Right	677	956	279	871	194

NOTES:

[1]Calculated horsepower is based on measured mast torque and 100% (324 RPM) rotor speed.
[2]During autorotation there is no drive torque and boot was not cycled for this flight condition

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